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TOPEX Satellite Option Study

Final Report

JPL CONTRACT NO. 956198, BASIC

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Prepared for:
Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California

Prepared by:
RCA Government Systems Division
Astro-Electronics, Princeton, New Jersey

R-4402
May 28, 1982

RCA Astro-Electronics

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INTRODUCTION AND SUMMARY

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INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

The purpose of the TOPEX Satellite Option Study is to assess the applicability of an existing spacecraft bus and subsystems to the requirements of ocean circulation measurements. RCA's recommended candidate is the operational meteorological satellite family TIROS and DMSP which we build respectively for the civilian agency, NOAA, and the military agency, Air Force. These programs utilize a common bus to satisfy their earth observation missions. Note that although the instrument complements were different, the pointing accuracies were different, and, initially, the boosters were different (DMSP used the Thor booster and TIROS the Atlas), a high degree of commonality was achieved. This experience was used in developing the TOPEX approach presented herein. During the 23 years we conducted the TIROS and then DMSP programs, we developed procedures and techniques for interface control, integration, and test of multiple payloads on earth observation satellites. This experience has led to low cost approaches resulting from evolutionary development. We utilized this to prepare technical and programmatic concepts for the TOPEX mission. In particular, we relied heavily on the heritage of the Advanced TIROS-N (ATN) program.

There have been other studies, relating to changes to TIROS and DMSP and relating to other applications of these buses, that were utilized in the TOPEX study. Specifically, extensive use was made of the design effort funded by NASA to configure ATN to be Atlas and Shuttle compatible. This is referred to as the Shuttle/Atlas ATN (SAATN) configuration.

We conducted similar studies on DMSP and are currently funded by the Air Force to design F-15 for a Shuttle launch. Other programs that bear on the TOPEX study are:

- Maritime Applications Experiment (MAE) funded by NASA to incorporate a Coastal Zone Color Scanner (CZCS), a Scatterometer (SCATT) and a Microwave Imager (SSM/I) on the NOAA-I[1] and -J spacecraft. These would be piggy-backed with the meteorological and climate sensors already planned for these flights.

- Navy Remote Oceanography Satellite System (NROSS) which we have studied under our own funds (and are hopeful for Navy support) to refurbish the NOAA-D bus (which exists but is not scheduled for launch) to carry SCATT, an SSM/I and an Advanced Very High Resolution Radiometer (AVHRR).

We have also investigated applications of TIROS-N, ATN, and DMSP to Earth Resource Missions, Synthetic Aperture Radar Missions, a Halley Comet Imaging Mission, and currently, under JPL direction, an Anteros Asteroid Mission. In all cases, we and our prospective users were interested in low cost application of proven bus technology: a bus, which due to its use in an operational mission must provide reliable, long-life operation.

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1. Each spacecraft in the current TIROS series is named NOAA-(Letter).

RCA Report R-4402 dated March 11, 1982 was reviewed by the TOPEX Satellite Options Team. In Letter No. 622-SPD:ja dated April 26, 1982, JPL requested answers to questions resulting from that review. All but six of the answers were provided to and discussed with JPL on May 7, 1982. We have included a complete set of questions and answers in Appendix A, herein.

1.2 TECHNICAL SUMMARY

Using the ATN configuration as a baseline, we developed concepts for all specified options. We included in our study the possibility of using the Shuttle (STS) or Delta boosters; the Atlas booster was added because it is the current TIROS and DMSP booster and would have minimum impact on the spacecraft bus design.

The ATN Spacecraft is shown in Figure 1-1. It represents an evolution from the original DMSP 5D-1 which led to TIROS-N[1] and then to ATN. Briefly, it is a meteorological spacecraft which is continuously pointed at nadir as the satellite circles the earth at 450 nmi in a sun synchronous orbit. The solar array is continuously oriented toward the sun by a solar array drive that is controlled by a central processing unit (CPU). The major structural elements of the spacecraft are:

- RSS (Reaction Control Equipment Support Structure) which carries the RCE, batteries, and upper stage solid motor, as well as interfacing with the Atlas.
- ESM (Equipment Support Module) which carries most of the support subsystem electronics, as well as payload elements.
- IMP (Instrument Mounting Platform) which provides a common structure for instruments which require coalignment.

The spacecraft subsystems are as follows:

- Structure Subsystem.
- Thermal Subsystem -- passive control with active augmentation.
- Attitude Determination and Control Subsystem (ADACS)--a zero momentum, four reaction wheel system using earth sensors and gyro-sun sensor's for measurement, and magnetics for external torquing. Designed on DMSP to provide 0.01° control and 0.2° on TIROS.
- Command and Data Handling Subsystem (CDHS)--a centralized computer (CPU) controlled system provides fault tolerant performance, with special-purpose processors for primary data and an array of five NASA standard DTR's for data storage.
- Communications Subsystem (CS)--includes downlinks at VHF, L-Band, and S-Band, and uplink commands at VHF.
- Propulsion Subsystem--the final stage of ascent and attitude control functions.
- Power Subsystem--the primary (solar-array) and secondary power (NiCd batteries) sources and associated regulation electronics which are characterized by self-checking automatic redundancy switching with ground commanded override.

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1. Reference "The TIROS-N/NOAA A-G Satellite Series", NOAA TM NESS 95, A. Schwalb, March 1978.

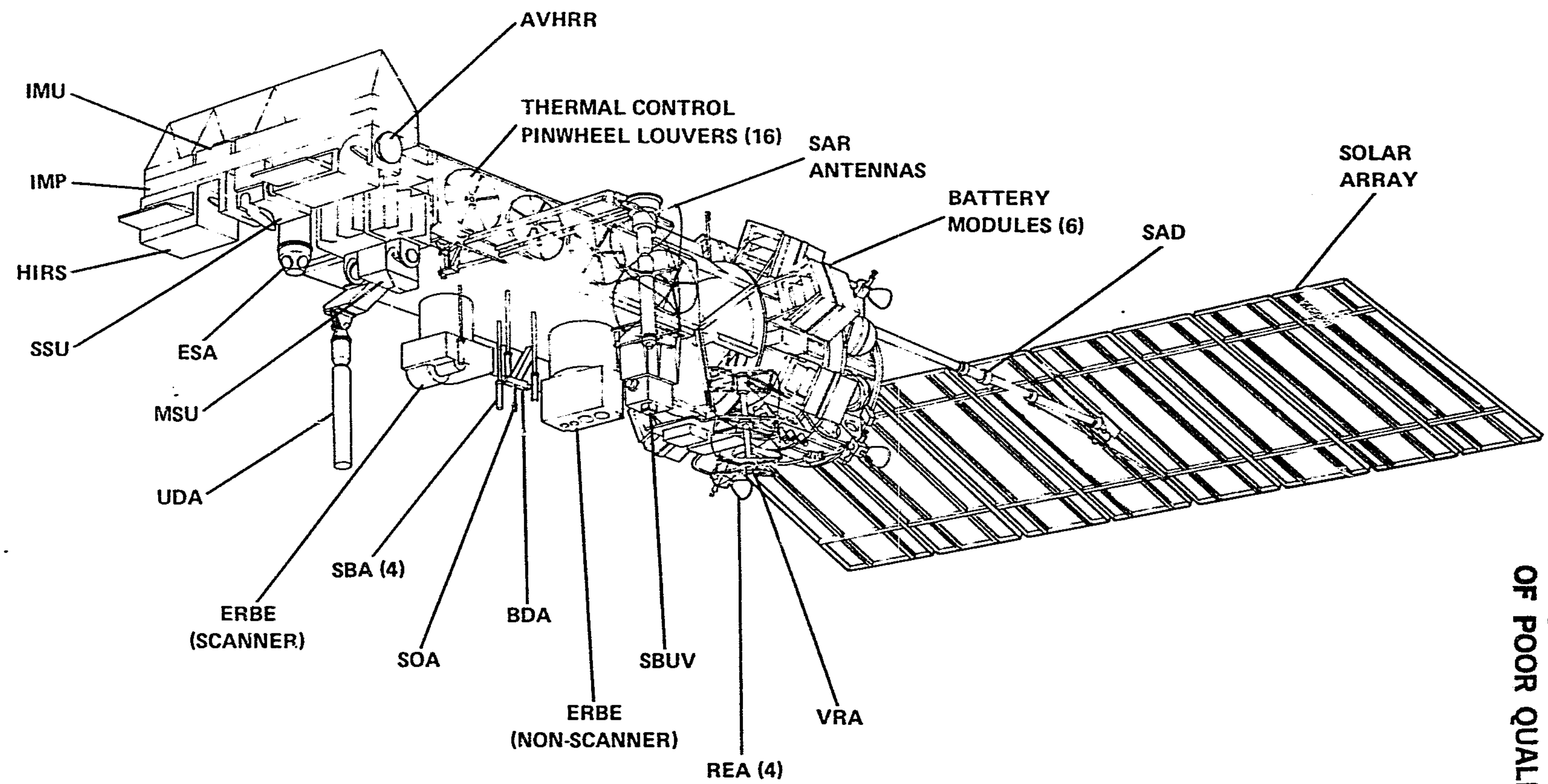

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Figure 1-1. Advanced TIROS-N Spacecraft, NOAA-G Configuration

The ATN spacecraft is designed with no single-point failures and with a significant level of automatic switching as described in the Power Subsystem. The central computer also does a variety of health checks on various subsystems and, should a major fault be detected, it will command the spacecraft to a "safe-state". This high degree of autonomy minimizes ground control requirements. Tape recorder management and clock updates are the primary ground command function.

This ATN configuration has been modified to provide concepts for each of the options. The SAATN effort was used to define those options corresponding to the STS launch. Details of all the configurations and payload mechanical accommodation are given in Section 3.0.

Figure 1-2 shows our concept for Option 1 stowed in the Orbiter bay. From technical considerations, the least complex configuration is Option 2 or 3 launched by a Delta. The least change to the spacecraft bus would be 2 or 3 launched by an Atlas. Option 1 is most readily accommodated by the STS. On Atlas or Delta, the 2-meter dish of Option 1 would be deployed, and concern exists for the maintenance of alignment and the complexity of the deployment device.

Details of the bus subsystem accommodation to the TOPEX options are given in Section 5.0. The ADACS, Power, Thermal, and C&DH Subsystems are unmodified or have minor modifications. The largest change to ATN for all the options is in the telecommunications area. The uplink and downlink hardware must be replaced by TDRS- and STDN-compatible equipment. Depending on the choice of boosters, the ascent propulsion must be changed, and there are some modifications to the structure.

The liftoff weight for selected TOPEX options is summarized in Table 1-1. Although the ascent propulsion system selected for the STS approach would have to be offloaded significantly (to approximately one fifth of its capacity), we elected to stay with these estimates anticipating that some compromise in the STS orbit will be made to avoid a dedicated Shuttle launch. Furthermore, the overall cost model would not change drastically with smaller propellent tanks. Notice that the Delta options easily satisfy the weight constraints of this booster. Finally, the AKM expendable weight on the Atlas option was maintained, although some change is expected (downward if direct ascent with a minimum dogleg maneuver can be achieved; alternatively, a less efficient Atlas trajectory could be flown as used on the first TIROS-N).

An equipment family tree for the TOPEX spacecraft is shown in Figure 1-3. A summary of equipment heritage, mass, quantity per spacecraft, and other pertinent information is provided in Table 1-2.

1.3 PROGRAMMATIC SUMMARY

A discussion of the assumptions which led to the schedule presented for TOPEX is given in Section 6.0. This is a 46-month schedule, as compared to our typical 36-month TIROS schedule. This was due to the assumption of an STS launch option. This schedule could be reduced by 4 months if a detailed design study phase preceded the kickoff of hardware. It will be reduced further if either TIROS or DMSP elect to launch by the Shuttle prior to the start of TOPEX. Several studies of Shuttle compatibility have been conducted by RCA for both the TIROS and DMSP programs.

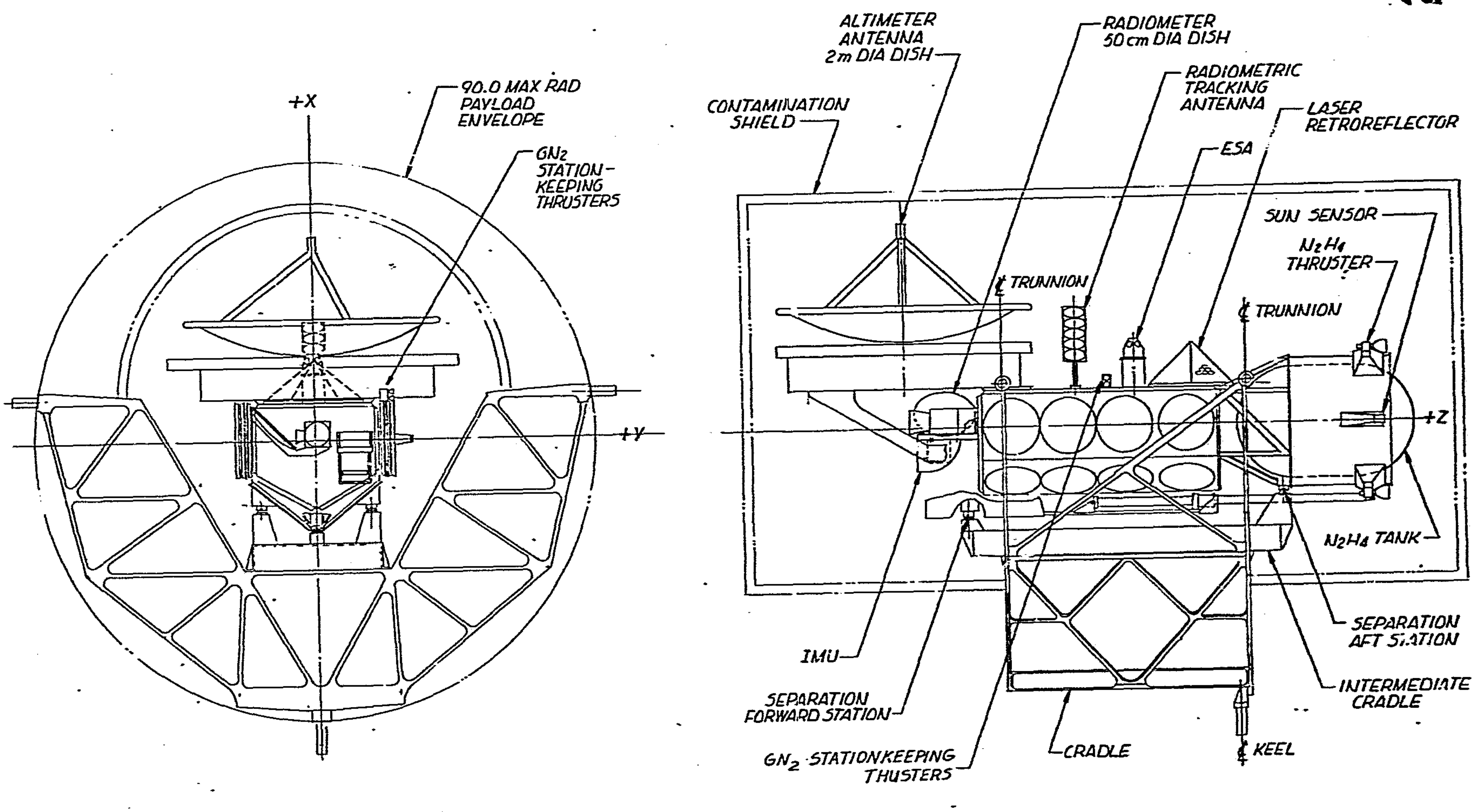

Figure 1-2. Option 1 in the Orbiter Bay

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TABLE 1-1. LIFTOFF WEIGHT FOR TOPEX OPTIONS

Item	STS			Delta		Atlas
	Option 1	Option 2	Option 3	Option 2	Option 3	Option 3
Payload	214.5	165.1	215.1	165.1	215.1	215.1
Communications	22.0	22.0	22.0	22.0	22.0	22.0
Harness	68.0	68.0	68.0	68.0	68.0	68.0
Power	154.0	154.0	154.0	154.0	154.0	154.0
DHS and C&C	71.4	71.4	71.4	71.4	71.4	71.4
Thermal	33.8	33.8	33.8	31.8	31.8	31.8
ADACS	35.5	35.5	35.5	35.5	35.5	35.5
AKM Case	0.0	0.0	0.0	0.0	0.0	0.0
Structure	204.5	204.5	204.5	160.0	160.0	173.0
Margin	22.7	22.7	22.7	22.7	22.7	13.6
RCE (Dry)	106.6	106.6	106.6	16.8	16.8	42.7
Balance	45.0	45.0	45.0	45.0	45.0	45.0
Spacecraft Dry	978.0	928.6	978.6	792.3	842.3	929.1
N_2H_4	329.7	233.5	192.0	0	0	56.0
GN_2	66.0	66.0	66.0	16.5	16.5	33.0
AKM Expendables	0.0	0.0	0.0	0.0	0.0	664.0
Total Spacecraft	1373.7	1228.1	1236.6	808.8	858.8	1673.1

1.4 COST

Under separate cover, we have provided cost estimates down to the subsystem WBS level for the TOPEX mission. We have included in the present volume the assumptions which led to the development of this data.

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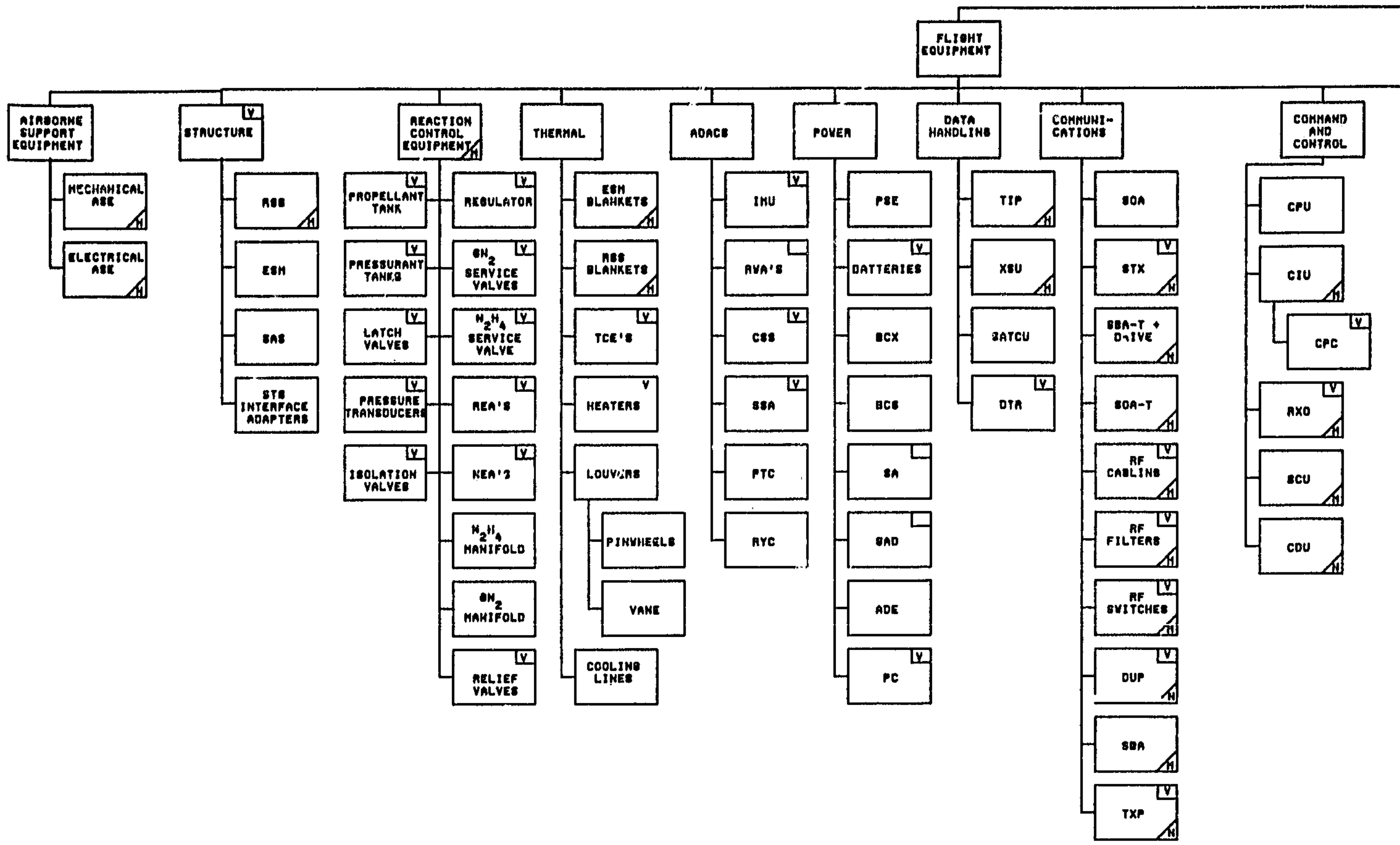

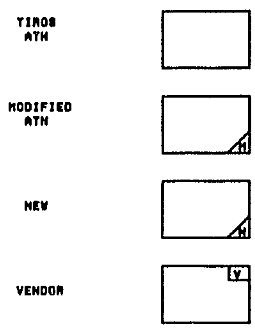

ADE	ARRAY DRIV
AGE	AEROSPACE
AOS	ASCENT GUI
ALT	ALTIMETER
BCS	BATTERY CU
BCX	BATTERY CHA
CDU	COMMAND DI
CIU	CONTROLS I
CMD	COMMAND
COMM	COMMUNICATI
CPC	CONTROLS P
CPU	CENTRAL PR
CSS	CONICAL SC
DTR	DIGITAL TA
DUP	DUPLEXER
EMI	ELECTROMAG
ESA	EARTH SENS
ESM	EQUIPMENT
GFE	GOVERNMENT
GPS RCVR	GPS SERIES

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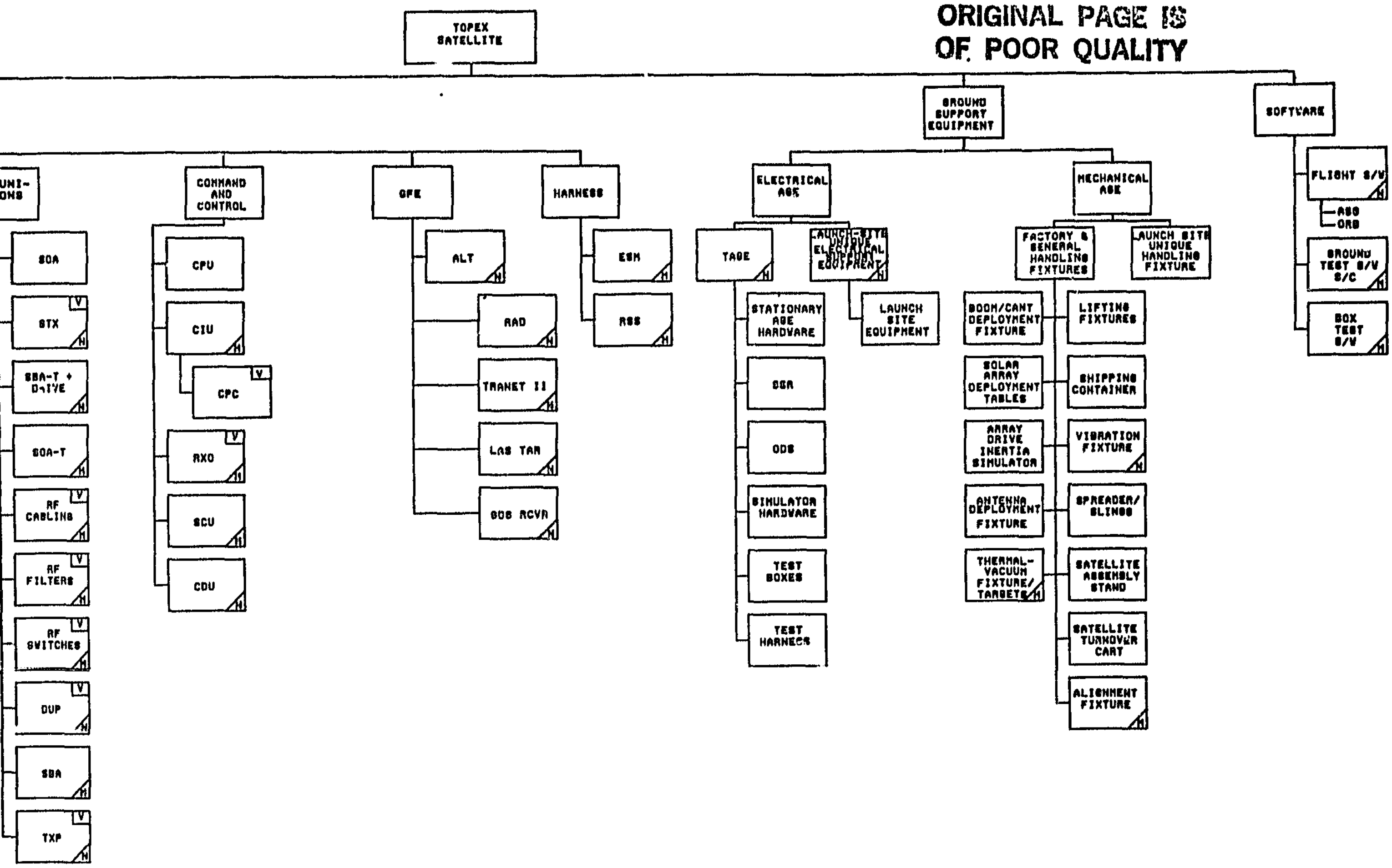

ACRONYMS					
ADE	ARRAY DRIVE ELECTRONICS	IMU	INERTIAL MEASUREMENT UNIT	SAS	SOLAR ARRAY SUPPORT
AGE	AEROSPACE GROUND EQUIPMENT	LAS TAR	LASER RANGING TARGET	SATCU	SOLAR ARRAY TELEMETRY COMMUTATOR UNIT
AGS	ASCENT GUIDANCE SOFTWARE	NEA	NITROGEN ENGINE ASSEMBLY	SDA	S-BAND ANTENNA (DIRECTIONAL)
ALT	ALTIMETER	ODS	ORDNANCE DEVICE SIMULATOR	SDAT	TDRSS HIGH GAIN
BCS	BATTERY CURRENT SENSOR	PC	POWER CONVERTER	SCU	SIGNAL CONDITIONING UNIT
BCX	BATTERY CHARGE ASSEMBLY EXPANDED	PSE	POWER SUPPLY ELECTRONICS	SOA	S-BAND OMNI ANTENNA
CDU	COMMAND DISTRIBUTION UNIT	PTC	PITCH TORQUE COIL	SOAT	S-BAND OMNI ANTENNA (TDRSS)
CIU	CONTROLS INTERFACE UNIT	RAD	MICROWAVE-RADIOMETER	SSA	SUN SENSOR ASSEMBLY
CMD	COMMAND	REA	REACTION ENGINE ASSEMBLY	SSR	SATELLITE SUPPORT RACK
COMM	COMMUNICATIONS	RCE	REACTION CONTROL EQUIPMENT	STX	S-BAND DATA TRANSMITTER
CPC	CONTROLS POWER CONVERTER	RF	RADIO FREQUENCY	TAGE	TOPEX AUTOMATIC GROUND EQUIPMENT
CPU	CENTRAL PROCESSING UNIT	RSS	RCE SUPPORT STRUCTURE	TCE	THERMAL CONTROL ELECTRONICS
CSS	CONICAL SCANNING SENSOR	RWA	REACTION WHEEL ASSEMBLY	TIP	TIROS-N INFORMATION PROCESSOR
DTR	DIGITAL TAPE RECORDER	RXO	REDUNDANT CRYSTAL OSCILLATOR	TLM	TELEMETRY TXP TDRSS TRANSPONDER
DUP	DUPLEXER	RYC	ROLL-YAW TORQUE COIL	TXP	TDRSS TRANSPONDER
EMI	ELECTROMAGNETIC INTERFERENCE	SA	SOLAR ARRAY	TRANET II	TRACKING/RANGING SYSTEM
ESA	EARTH SENSOR	SAD	SOLAR ARRAY DRIVE	XCU	CROSS-STRAP UNIT
ESM	EQUIPMENT SUPPORT MODULE				
GFE	GOVERNMENT FURNISHED EQUIPMENT				
GPS RCVR	GPS SERIES RECEIVER				

Figure 1-3. TOPEX Spacecraft Equipment Family Tree, Shuttle Launched Option

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TABLE 1-2. EQUIPMENT HERITAGE AND DESCRIPTION (STS Option 1)

System/Assembly	Project Heredity	Design Heredity	Mass (kg)	Qty/ S/C	Redundancy	Spares[1]	Eng. Model	Qual. Model
Power								
PSE	ATN	100%	11.8	1	100% elec	1	0	0
Batteries	ATN	100%	74.3	3	3 for 2	1	0	0
BCX	ATN	100%	6.5	1	100% elec	1	0	0
BCS	ATN	100%	1.4	3	100% elec	1	0	0
SA	ATN	100%	52.0	1	2 yr design rqmt.	0	0	0
SAD	ATN	100%	4.6	1	100% elec	0	0	0
ADE	ATN	100%	2.6	1	100% elec	1	0	0
PC	ATN	100%	0.5	1	100% elec	1	0	0
Data Handling & Command								
TIP	ATN	75%	7.3	1	100% elec	1	0	0
XSU	ATN	75%	3.8	1	100% elec	1	0	0
SATCU	ATN	100%	0.2	1	Not mission critical	0	0	0
TR	ATN	100%	30.5	3	3 for 2 + 100% redun.	1	0	0
CPU	DMSP	100%	11.0	2	100% elec	1	0	0
CIU	ATN	95%	9.7	1	100% elec	1	0	0
CPC	ATN	100%		1	100% elec	1	0	0
RXO	Nova	80%	1.1	2	100% elec	1	0	0
SCU	ATN	90%	3.2	1	100% elec	1	0	0
CDU	TIROS	30%	4.5	1	100% elec	1	0	0
Structure								
RSS	ATN/SAATN	50% (ATN)[2]	36.3	1	NA	0	0	0
ESM	ATN/SAATN	80% (ATN)[2]	83.9	1	NA	0	0	0
SAS	ATN/SAATN	75% (ATN)[2]	13.8	1	NA	0	0	0
STS Interface Adapt	ATN/SAATN	0 (ATN)[2]	31.0	1	NA	0	0	0
Misc.	ATN/SAATN	75% (ATN)[2]	39.5	1	NA	0	0	0
Propulsion								
N_2H_4 Tank	Viking	100%	43.2	1	NA	Partial	0	0
GN_2 Tank	GPS	100%	39.6	6	Ascent-no miss-100%	1	0	0
REA	Voyager	100%	6.1	4	100%	1	0	0
NEA (2 lb_f)	ATN, DMSP	100%	1.5	8	33%	2	0	0
NEA (0.2 lb_f)	Classified	100%	1.4	4	100%	1	0	0
Misc.	All flown	100%	13.9	1 set	NA	Partial	0	0

1. All vendor units will be spared as complete units; all RCA units will be spared at the board level.
2. 100% (SAATN).

TABLE 1-2. EQUIPMENT HERITAGE AND DESCRIPTION (STS Option 1) (Continued)

System/Assembly	Project Heredity	Design Heredity	Mass (kg)	Qty/ S/C	Redundancy	Spares[1]	Eng. Model	Qual. Model
Thermal								
ESM Blankets	ATN	95%		1 set	NA-design margin	0	1	0
RSS Blankets	ATN	75%	13.8	1 set	NA-design margin	0	1	0
TCE's	ATN	100%	3.1	37		2	0	0
Heaters/ Radiators	ATN	100%	7.2	1 set	Passive	2	0	0
Louvers								
Pinwheels	ATN	100%	5.4	16		1	0	0
Vane	ATN	100%	1.8	12		1 set	0	0
Cooling Lines	ATN	100%	1.1	1 set	NA-Grd Test	NA		
RCE System	ATN	100%	1.4	1 set	NA	0	0	0
ADACS								
IMU	ATN	75%	10.4	1	4 for 3 gyros	1	0	0
RWA	ATN	75%	15.8	4	4 for 3	1	0	0
SSA	ATN	100%	0.9	1	Not mission critical	1	0	0
PTC	ATN	100%	1.1	1	100% elec	0	0	0
RYC	ATN	100%	2.2	1	100% elec	0	0	0
CSS	ATN/SAATN	0% (ATN)[2]	5.1	2	100% elec	1	0	0
Telecommunications								
TDRSS XPON	DE	100%	7.0	2	100% elec	1	0	0
TDRSS H-GA	SMM	100%	11.32	1	100% elec	1	0	0
TDRSS OMA	ATN	60%	0.05	1	NA-backup	0	0	0
STX	ATN	80%	1.0	1	NA-backup	0	0	0
SBHGA	ATN	90%	0.08	1	NA-backup	0	0	0
SBOMA	ATN	90%	0.05	2	NA-backup	0	0	0
Misc RF	Various	50%	2.5	1 set	NA-passive elements	0	0	0
Payload								
Altimeter			94/74/94			0	0	0
Radiometer			18.5			0	0	0
Laser-Retro			25/13/6			0	0	0
Tranet II			53.0			0	0	0
GPS			30.0			0	0	0
Laser Cal TGT			13.6					
Harness	ATN	50%	68.0	1	NA			
ASE								
Mechanical	SAATN	0% (ATN)[2]	650.0	1	NA	0		
Electrical	SAATN	0% (ATN)[2]	15.0	1	100% elec	1		

1. All vendor units will be spared as complete units; all RCA units will be spared at the board level.
2. 100% (SAATN).

SECTION 2.0

REQUIREMENTS SUMMARY

SECTION 2.0
REQUIREMENTS SUMMARY

2.1 MISSION DESCRIPTION

The main thrust of the TOPEX mission is to build on the experience gained by the GEOS-3 and SEASAT missions which indicated that precise studies of ocean topography can be done via satellite altimetry. This topographical information, when acquired over several years from a repetitive orbit cycle, can provide a valuable new insight to global ocean dynamics, specifically in the critical areas of general circulation and its variation, the relation of low surface winds to circulation, and tidal energy dissipation. In addition, the TOPEX data will provide a valuable tool for coastal oceanography (by providing sea height in coastal regions) and will provide improved geoid measurements, an area of broad geophysical interest.

In order to address these global topics, large numbers of measurements with complete global coverage on a relatively short time scale is required. This implies a satellite borne instrument complement capable of remote surface and/or near surface measurements must be employed. The general observational scenario and its associated spacecraft requirements are conceptually similar to those required for meteorological observations by the low, polar orbiting meteorological satellite TIROS; therefore, it is not surprising that the TIROS technology provides a good conceptual match for the TOPEX mission (a comparison of the TOPEX and TIROS payloads is in Section 2.2).

Since the objective of the mission is to measure the sea height with respect to a known reference, we need to know (1) exactly where the satellite is with respect to a known reference and (2) the precise altitude of the spacecraft over the ocean. These requirements can be met by an instrument complement consisting of an altimeter (the primary sensor for the altitude measurement), a radiometer (required for path length connections to the altimeter data due to water vapor in the troposphere), and some accurate means of spacecraft tracking (e.g., laser ranging, Tranet II, and/or GPS in some combination).

In addition, surface winds must be measured in order to understand their relationship to ocean circulation. While this is not required to be done from the TOPEX spacecraft per se (i.e., it is not necessary for data reduction), it is imperative that global, low altitude wind data be available from some source for the regions and times of the TOPEX data (i.e., it is necessary for data analysis). As a consequence, the option of including a scatterometer (for surface wind measurements) on the TOPEX spacecraft may become necessary to provide this observation if it is not available from some other project. Although this possibility was not considered in the study, we have done so for the Maritime Applications Experiment (MAE) option for NOAA-J. In that study, the spacecraft had the full manifest of meteorological sensors in addition to the scatterometer. We found no significant problems in accommodating the scatterometer on the TIROS spacecraft. Should this requirement arise for TOPEX, a similar study would be required to verify the conclusion.

Given the instrument selection, the spacecraft orbit selection options become quantifiable. The circular orbits selected for the three options are shown in Table 2-1. Booster capabilities and propulsion requirements are also given. Performance calculations for the Atlas booster into the TOPEX orbit options have not been completed as yet.

TABLE 2-1. MISSION OPTIONS

Option	Operational* Altitude (km)	ΔV SSV Orbit** (m/s)	Delta Direct Injection Capability†		Stationkeeping ΔV†† m/s/5 years
			3910	3920	
1	1334	550	1570	1970	Negligible
2	1000	390	1750	2140	0.73
3	800	290	1870	2270	2.33

*63.4° Inclination

**63.4° 280 km SSV orbit assumed

†ELV launches inject directly into operational altitude and inclination, no dogleg

††Air drag only

The tradeoff in altitude between the three options is primarily driven by orbit stability and knowledge versus altimeter requirements. The differences between the baseline instrument design for the three orbits reflects this tradeoff; i.e., the highest orbit requires a bigger, more powerful dual-frequency altimeter, but simpler tracking.

2.2 PAYLOADS ACCOMMODATION

The instrument accommodation requirements for all three options are summarized in Table 2-2. The table is structured by measurement type for each option. The altimeter for Option 1, the high altitude case, poses the most stringent attitude control requirements, highest power requirements, and deployment requirements for ELV launches, but it is still within the capabilities of the TIROS bus as shown in Table 2-3. The requirement for the larger dish (2 meter) is more easily accommodated with minimum change and risk with the STS launch option. With the ELV option, the dish would have to be deployed. This creates concern for the deployment mechanism reliability and the attainment of alignment. The field-of-view requirements for the altimeter can be satisfied in any option.

The major radiometer constraint is the location of the skyhorn to ensure that deep-space-viewing requirements are met. The layouts to accommodate this requirement are given in Section 3.0.

TABLE 2-2. SUMMARY OF TOPEX INSTRUMENTS

Radiometric Tracking		
Parameter	Options 1 and 2 Laser/Tranet II	Option 3 Tranet II/Series GPS
Mass	53 kg	83 kg
Power	40 W	60 W
Antenna Type	Double helix	Omni/double helix
Pointing	Nadir ±5°	n/A(up)/nadir ±5°
FOV	60° cone nadir	100° cone(up)/60° cone nadir

Laser Retroreflector			
Parameter	Option 1 Laser Prime	Option 2 Laser Backup	Option 3 Laser Calibration
Mass	25 kg	13.6 kg	13.6 kg
Pointing	50° Apex Half angle pyramid 70 cm Base Nadir TBD	1.15 m torus Around altimeter Antenna Nadir TBD	1.15 m torus Around altimeter antenna Nadir TBD
FOV Half Angle	55° cone angle Nadir	55° cone angle Nadir	55° cone angle Nadir

TABLE 2-2. SUMMARY OF TOPEX INSTRUMENTS (Continued)

Altimeters			
Parameter	Option 1 TOPEX Dual Frequency Altimeter	Option 2 Improved SEASAT Altimeter	Option 3 TOPEX Ku-Band Altimeter
Mass	94 kg (w/o ant)	74 kg (w/o ant)	94 kg (w/o ant.)
Power	199 W	120 W	120 W
Output	13.7 GHz/5.45 GHz	13.5 GHz	13.7 GHz
Bandwidth	2.34-600 MHz	320 MHz	2.34-600 MHz
Antenna Type	2 m parabolic	1 m parabolic	1 m parabolic
Pointing	Nadir ±0.15	Nadir ±0.25°	Nadir ±0.25°
Knowledge	Nadir ±0.05	Nadir ±0.10°	Nadir +0.10°

Radiometers	
Parameter	Options 1, 2, and 3 TOPEX 2-Channel Radiometer
Mass	18.5 kg (w/o ant)
Power	20 W
Output	20.3/31.4 GHz
Antenna	50 cm Hyperbolic, Skyhorn
Pointing	Altimeter boresite ±0.25°, Skyhorn normal to sun line
FOV	10° cone angle

TABLE 2-3. SUMMARY OF TOPEX PAYLOAD REQUIREMENTS (worst case option for all parameters) AND TIROS CAPABILITIES

Parameter	TOPEX Requirements	TIROS Capabilities (ATN)
Instrument Mass	214 kg (Option 3)	344 kg
Instrument Power	259 W (Option 1)	290 W
Instrument Pointing	Nadir $\pm 0.15°$ (Option 3)	Nadir ± 0.20 (spec 3σ)
Pointing Knowledge	Nadir $\pm 0.05°$ (Option 3)	Nadir ± 0.10 (spec 3σ)
Data Rate (Real Time)	$\leq$ 14.5 kbps	665.4 kbps
Uplink Rate	1 kbps	1 kbps
On-Board Storage	3.6×10^8 bits	4.5×10^8 bits/ transport (10 transports)
Oscillator Stability	1 part in 10^{10}/day	1 part in 10^8/day

The instruments and devices required to support the tracking requirements, while very different for the three options, pose no significant problems in accommodation in any options.

In general, the specified TOPEX payload and launch vehicle options can be accommodated by the TIROS spacecraft for any of the TOPEX options with minor modifications and impacts on the spacecraft subsystems. Option 1 is clearly the more difficult of the three, especially when combined with the ELV option, but even in this case solutions exist. Options 2, 3, and 1 (with the SSV) are more straight-forward.

2.3 ATTITUDE DETERMINATION AND CONTROL

Since the major objectives of the TOPEX mission require altimetry, precise attitude and orbit control are required to accommodate the sensors. The altimeter is the driving requirement (see Table 2-2), and the control/knowledge requirements for the altimeter antenna are between 0.25°/0.1° (Options 2 and 3) and 0.15°/0.05° (Option 1). The performance of the attitude determination and control system is described in Section 5.1.2.

The TIROS solar array is maintained in a sun-pointing position by computer control of the solar array drive; this mechanism will be retained unchanged for TOPEX. In general, the attitude disturbances for TOPEX will be less severe than for TIROS because there are fewer moving devices; this implies the RWA assembly should be adequate.

Finally, the TDRS requirement implies a high gain, pointable antenna be used which is capable of tracking TDRS during underflights. The TDRS antenna tracking will be done in two axes. The stability and pointing requirements imposed on the spacecraft by the payload instruments envelope are requirements imposed by the TDRS antenna. Its beamwidth, ~1° to 3°, is sufficiently large that errors in pointing (i.e. errors in the knowledge of exactly where we are in the track due to attitude and/or timing uncertainties) are small. This implies that the uncertainty in acquiring TDRS will be small and that no special attitude sensors or control elements (other than the development of the antenna control software) are required.

2.4 COMMAND AND DATA HANDLING

The Command and Data Handling (C&DH) subsystem for the TOPEX spacecraft will interact with all of the spacecraft subsystems, perform guidance and control calculations for spacecraft attitude and orbit attainment and maintenance, receive all commands and perform command decoding and distribution, and perform all data acquisition from the instruments. The basic requirements for the C&DH subsystem are summarized in Table 2-4.

2.5 TELECOMMUNICATIONS

The TOPEX spacecraft will communicate via TDRS (with the option of direct communications to the ground) via an S-Band link. The command uplinks will also be via TDRS with the same option. Both MA and SA links to TDRS are required. The baseline system is specified with an uplink frequency of 2.10740625 GHz and a downlink of 2.2875 GHz with a BER $<10^{-5}$. These are different frequencies, both up and down, to those on TIROS. Also the TDRS requirement would be new to TIROS. Of all the TIROS subsystems, this would be the most significantly modified. Fortunately, there exists qualified equipment from other sources to satisfy the TOPEX requirements.

The baseline TOPEX telecommunications subsystem has five functional antenna requirements: an articulated, two axis steerable, uplink gain dish for TDRS downlinks; a TDRS omni for uplink and emergency backup for downlink engineering data; an earth coverage antenna for ground-TOPEX communications; an earth omni for command; and one for engineering-telemetry.

2.6 PROPULSION

Depending on the launch vehicle option, the on-board propulsion system of the TOPEX spacecraft must provide ascent phase ΔV requirements, injection error correction, on-orbit stationkeeping, and attitude control. The broad requirements of the various orbit/booster options are summarized in Table 2-1.

As shown in the table for the Delta (ELV) case, a direct injection mode is used, thus there are no appreciable ΔV requirements on the spacecraft propulsion system, only insertion errors of the Delta and orbit maintenance. For the STS launches, the SAATN orbit insertion hydrazine system will be assumed. This system is sized to deliver a total 1040 m/sec ΔV to the ATN configuration; therefore, it easily meets all TOPEX requirements.

TABLE 2-4. C&DH SUMMARY

Function	Requirements
Uplink Commands	• 1-kbps uplink rate • Receive, decode, buffer, and distribute commands • Digital and serial commands required
Command Storage	• 512 stored commands required • 2 second command timer resolution (72 hour clock)
Systems Control/Safety/Monitoring	• Sequence and distribution of commands • Emergency procedures resident in spacecraft computer (interrupt driven) • Support spacecraft subsystems computer requirement • Provide an ultra-stable oscillator (< 1 part in 10^{10} day drift)
Data Acquisition	• A/D converter for science/ engineering functions • Analog subcommutator multiplexer capability • Serial digital data acquisition • Provide time sync and tag for all data and systems
Data Output (to Communications Subsystem))	• 1-2 kbps fixed format house-keeping data stream • 14.5-kbps real-time science data stream (record-in-parallel capability) • Format generation for real-time, tape record, and playback modes (14.5 kbps, 48 kbps, and 500 kbps rates)
Data Storage	• Provide on-board storage of $\geq 3.6 \times 10^8$ bits

The on-orbit ΔV adjust requirements for orbit maintenance derived from orbit decay will necessitate the addition of a separate propulsion system to be located in the ESM, with thrusters along the mission mode velocity vector. With a mass/drag ratio of 0.05 to 0.15 acceptable with the spacecraft long axis perpendicular to the orbit plane, the on-orbit orientation of the TOPEX spacecraft can be the same as TIROS. This will minimize spacecraft changes, but does introduce the need for the additional ESM propulsion system.

2.7 LAUNCH VEHICLE COMPATIBILITY

The TOPEX spacecraft design is based on ATN and incorporates elements proposed for the SAATN spacecraft for the STS launch option. The SAATN analysis considered those areas of unique concern to the STS (i.e., landing loads, safety requirements, and the like) and concluded that the ATN spacecraft can be SSV qualified in all areas with only minor modifications from the ELV version. This analysis carries over directly to TOPEX as does the ASE design concept.

The cradle can be used essentially as proposed for SAATN, although some minor payload driven modifications may be desirable. The preliminary analysis for TOPEX raises some question as to whether the cocoon proposed for SAATN is required for TOPEX. The cocoon was required for both contamination and thermal control in the STS bay. The decision to employ the cocoon for SAATN was ultimately driven by instrument contamination concerns which are not as significant for TOPEX. However, even if contamination was not a concern, thermal control may require the cocoon. This will depend on the details of the mission timeline and injection scenario, and will require further study before a definitive answer can be given. The command and data link through the SSV, TDRS, or directly to ground stations presents no unique problems.

For the ELV option, the major constraint is the shroud clearance. As discussed in Section 2.2, the 2-m altimeter antenna cannot be accommodated in the Delta shroud without requiring a deployment after injection. With this single caveat, the 86 inch x 157 inch dynamic envelope available for the Delta is adequate for the TIROS-based TOPEX spacecraft.

SECTION 3.0
CONFIGURATION OPTIONS

SECTION 3.0
CONFIGURATION OPTIONS

3.1 HERITAGE AND COMMONALITY

The TOPEX spacecraft proposed in this study is based on the flight-proven TIROS/DMSP integrated spacecraft system (ISS). Since it is an ISS system, the second or final stage of the launch vehicle is an integral part of the satellite. All versions of this design which have flown to date use an expendable launch vehicle (ELV) for the first stage and contain a TE-M-364-15 solid rocket motor as the apogee kick stage. The current design has evolved from the initial Thor-compatible DMSP to the current Advanced TIROS-N/Atlas configuration.

A 1980 NASA-funded study identified those modifications to the existing ATN design necessary to produce an STS-compatible spacecraft. The major changes were 1) the requirement for ascent support equipment to interface the spacecraft to the STS, 2) the replacement of the SRM with an all-liquid monopropellant propulsion system, and 3) the addition of structural mounting points to permit horizontal mounting of the spacecraft. This configuration was called SAATN.

For a Delta launch (proposed as a candidate booster for TOPEX), there would be no need for a high thrust capability on board the satellite, and that portion of the satellite that carries the solid motor (Atlas version) or hydrazine tank (STS version) can be reduced in size appreciably. This reduction allows the bus, fitted with TOPEX instruments, to fit comfortably in the shorter Delta fairing.

3.2 TOPEX OPTION 1

The Atlas launch vehicle, having a fairing with a 74-inch diameter dynamic envelope, cannot accommodate Option 1 of the TOPEX mission with its 2-meter altimeter antenna. We have not investigated the larger Atlas fairing. Figure 3-1 shows this option in a Delta launch configuration. The lower section, or adapter section, will attach to the Delta via a Marmon interface. Essentially the same complement of equipment now found on a TIROS or DMSP will be mounted on the adapter, viz: a GN_2 propulsion system for stationkeeping, batteries, charge amplifiers, and a deployment hinge for the solar array. For TOPEX Option 1, a sun sensor and a TDRSS antenna, gimballed for full hemispheric coverage, will also be mounted on the adapter section.

Forward of the adapter section is the equipment support module (ESM), a five-sided enclosure that houses the bulk of the satellite's electronic packages. The largest of the sides of the ESM is the earth-facing panel on which will be mounted the earth sensor assembly (ESA) or conical scan sensor (CSS), earth-pointing communication and tracking antennas, and the pyramidal laser retroreflector. The remaining four ESM panels contain thermal louvers on their external surfaces.

The forward end of the ESM will carry the half-meter radiometer dish and the 2-meter altimeter antenna. The altimeter dish is supported on a truss structure such that it is stowed concentric with the fairing while in the launch configuration.

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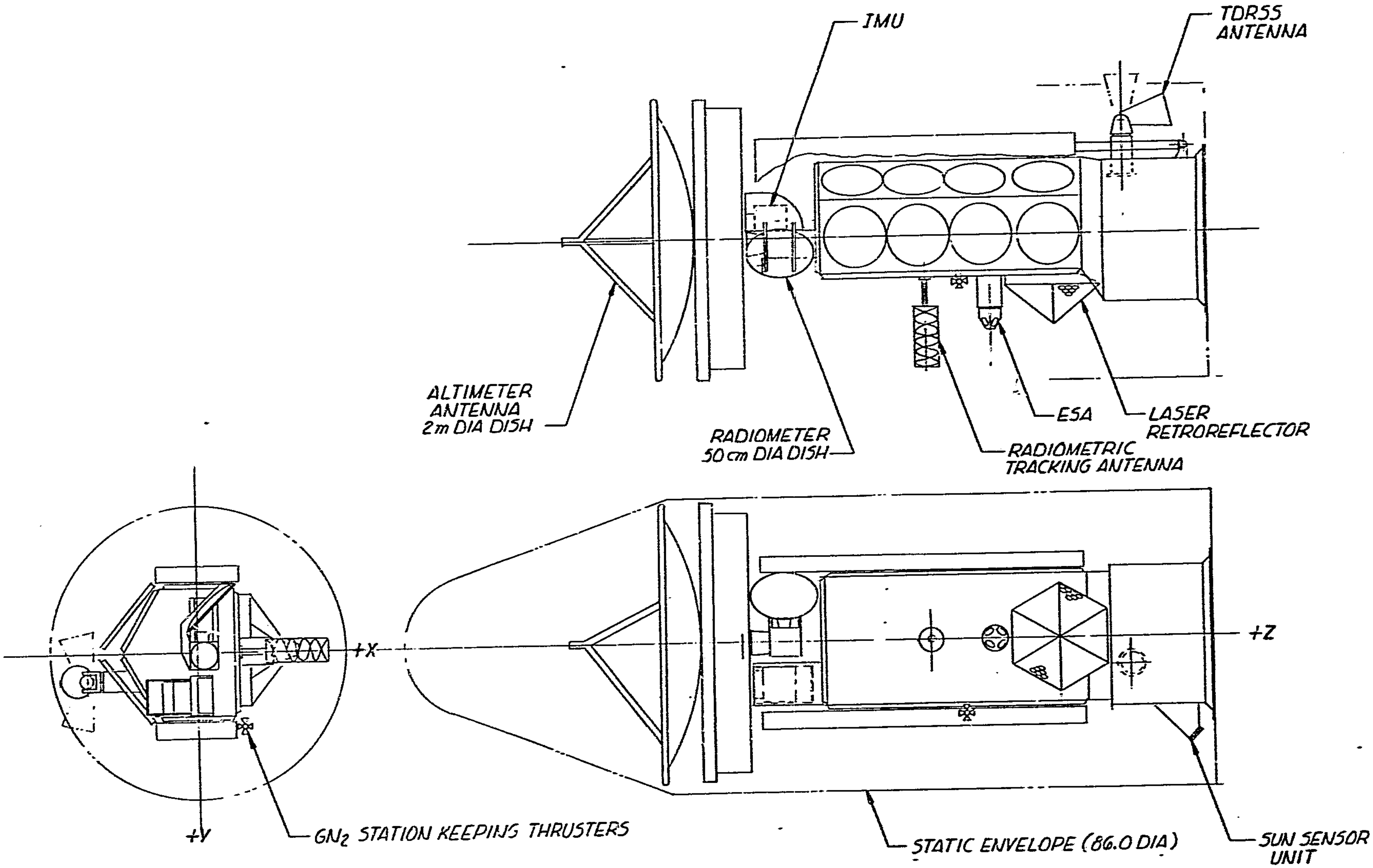

Figure 3-1. TOPEX Option 1, Delta ELV

The solar array is supported by a boom that is hinged near the aft end of the adapter section. In the launch configuration, the boom extends forward, parallel with the satellite's long axis. The array, hinged from the boom, folds around the body of the satellite and is held there with encircling restraint cables.

After orbit injection, the array restraint cables are cut, the panels unfold forming a 124-square-foot planar array, and the boom rotates 180° around its hinge (see Figure 3-2). The boom is equipped with a cant hinge that tilts the plane of the array for optimum sun incidence, plus a rotary drive that keeps the array normal aligned with the sun.

With the array deployed, a pyro-actuated release will allow the 2-meter altimeter dish to rotate into an earth-facing position where it is captured and latched. The 50-cm radiometer dish will similarly swing around to get a clear view of earth, and its cooling cone will extend to obtain an unobstructed view of space.

Option 1 for an STS launch is shown in Figures 3-3 (launch configuration) and 3-4 (on-orbit). Here the instrument arrangement differs in that both the altimeter and radiometer dishes are fixed in their orbit positions, thus requiring no deployment. The solar array, however, is stowed and will deploy in the same manner as described for the Delta launch.

In the Shuttle bay, the satellite is oriented with its long axis parallel with the Shuttle's, and the earth panel facing up (Shuttle +Z axis). It is mounted to an intermediate cradle via three explosive bolts, two on the adapter section and one at the forward end of the ESM. The intermediate cradle, which also serves as a handling and shipping pallet, is hard mounted to a PAM-D type cradle which connects to the Shuttle via standard trunion and keel fittings.

A contamination barrier, or "cocoon", surrounds the entire satellite while in the launch configuration. The lower half of this is of a rigid frame and panel construction attached to the cradle. The upper half retracts side-to-side, baby carriage fashion, to permit satellite ejection. The cocoon has appropriate ports and filters for purging and venting.

All of the explosive releases for the satellite and cocoon have redundancy. In the event of an abort, the cocoon can be commanded closed and latched to protect the satellite from contamination during landing.

3.3 TOPEX OPTIONS 2 AND 3

TOPEX Options 2 and 3 are similar, and are shown in Figures 3-5 and 3-6 (launch configuration) and Figure 3-7 (on-orbit configuration).

With the 1-meter altimeter dish, a common configuration will serve for an Atlas, Delta, or STS launch, with only minor differences as dictated by the unique interface requirements of each. The Delta and STS interfaces will be as described under Option 1, and for Atlas, a conical section with a Marmon interface adapts the aft section of the satellite to the booster.

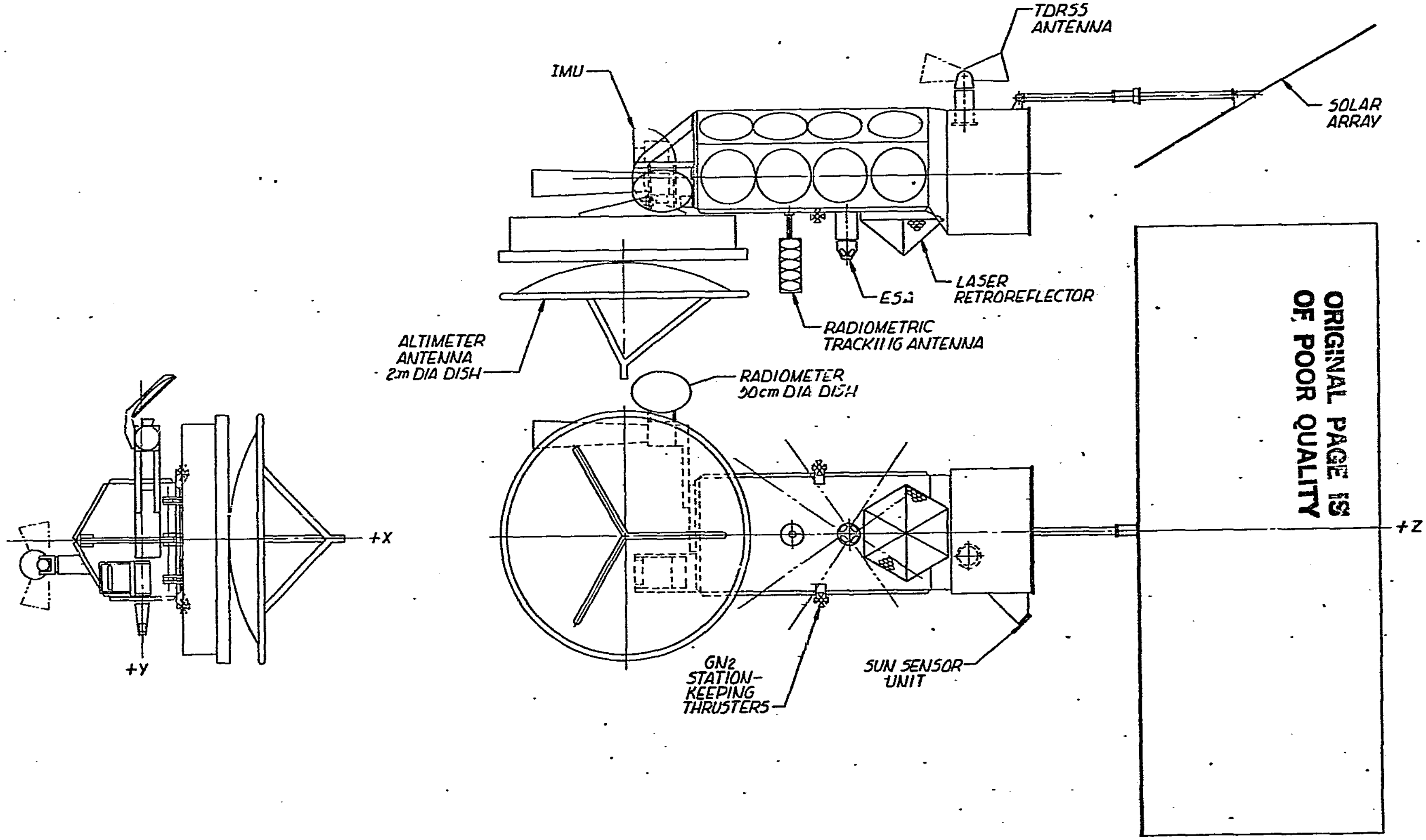

Figure 3-2. TOPEX Option 1, Delta, Orbit Configuration

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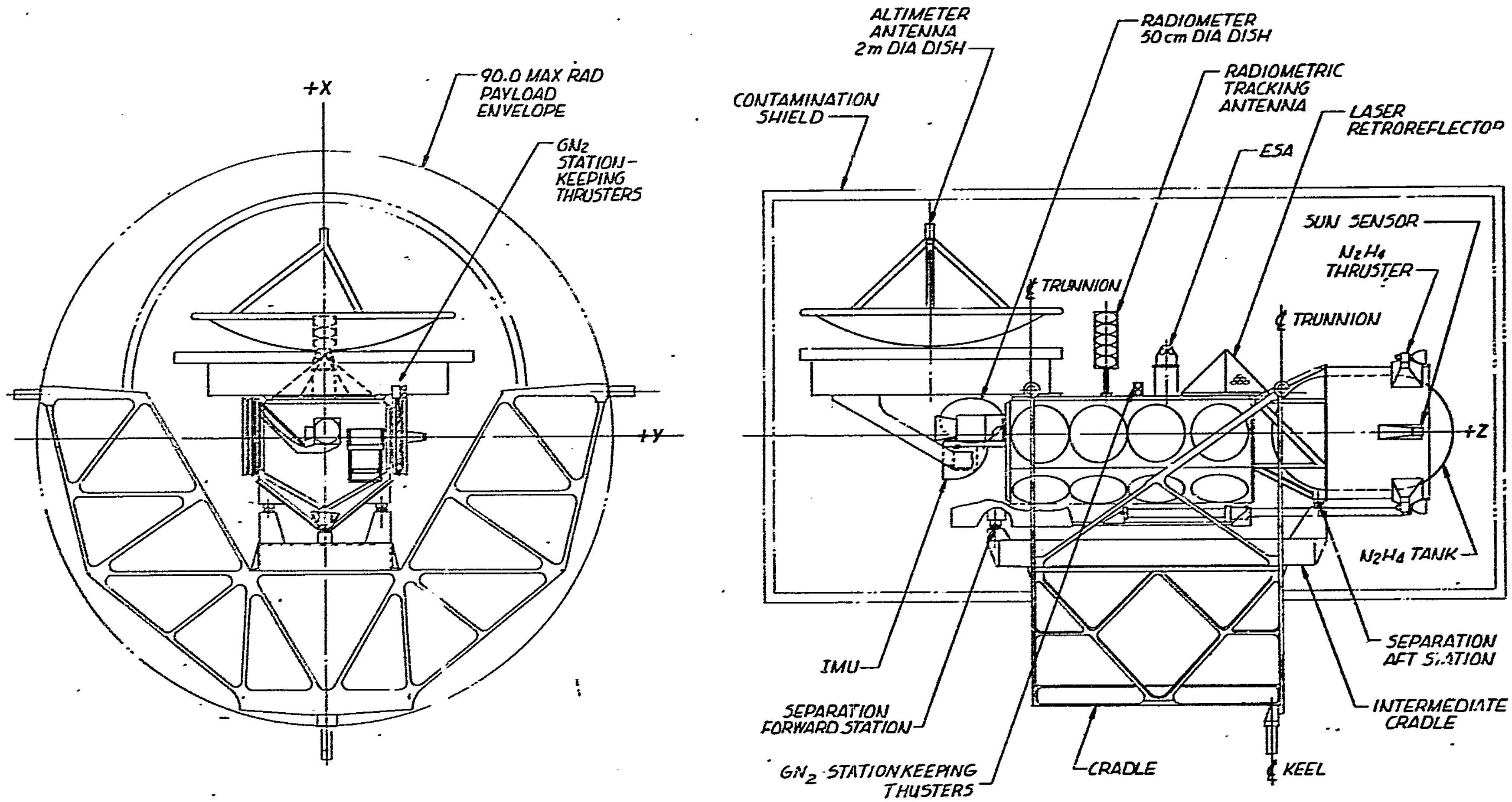

F 3-3. TOPEX Option 1, STS, Launch Configuration

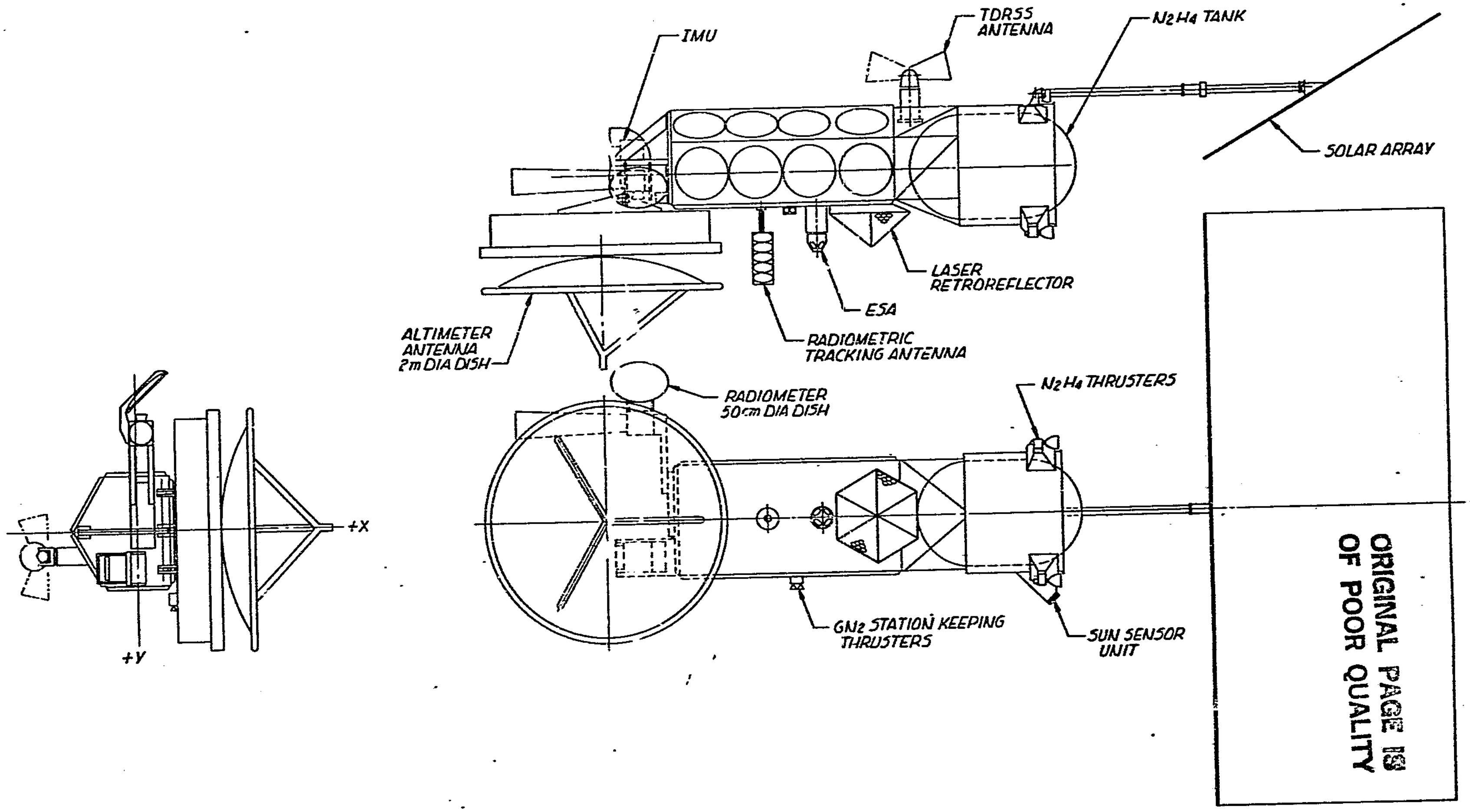

Figure 3-4. TOPEX Option 1, STS, Orbit Configuration

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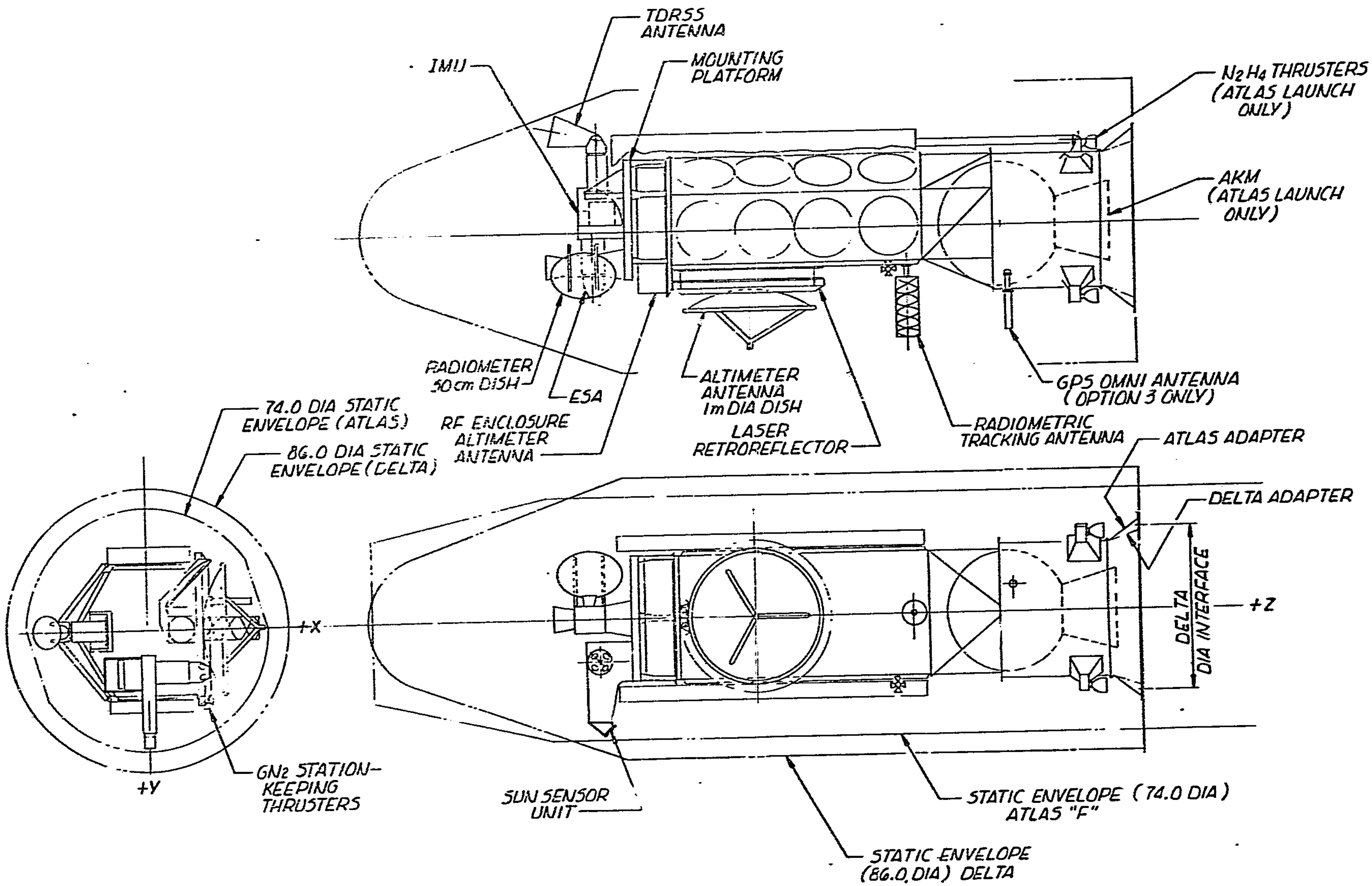

Figure 3-5. TOPEX Options 2 and 3, Delta and Atlas ELV, Launch Configuration

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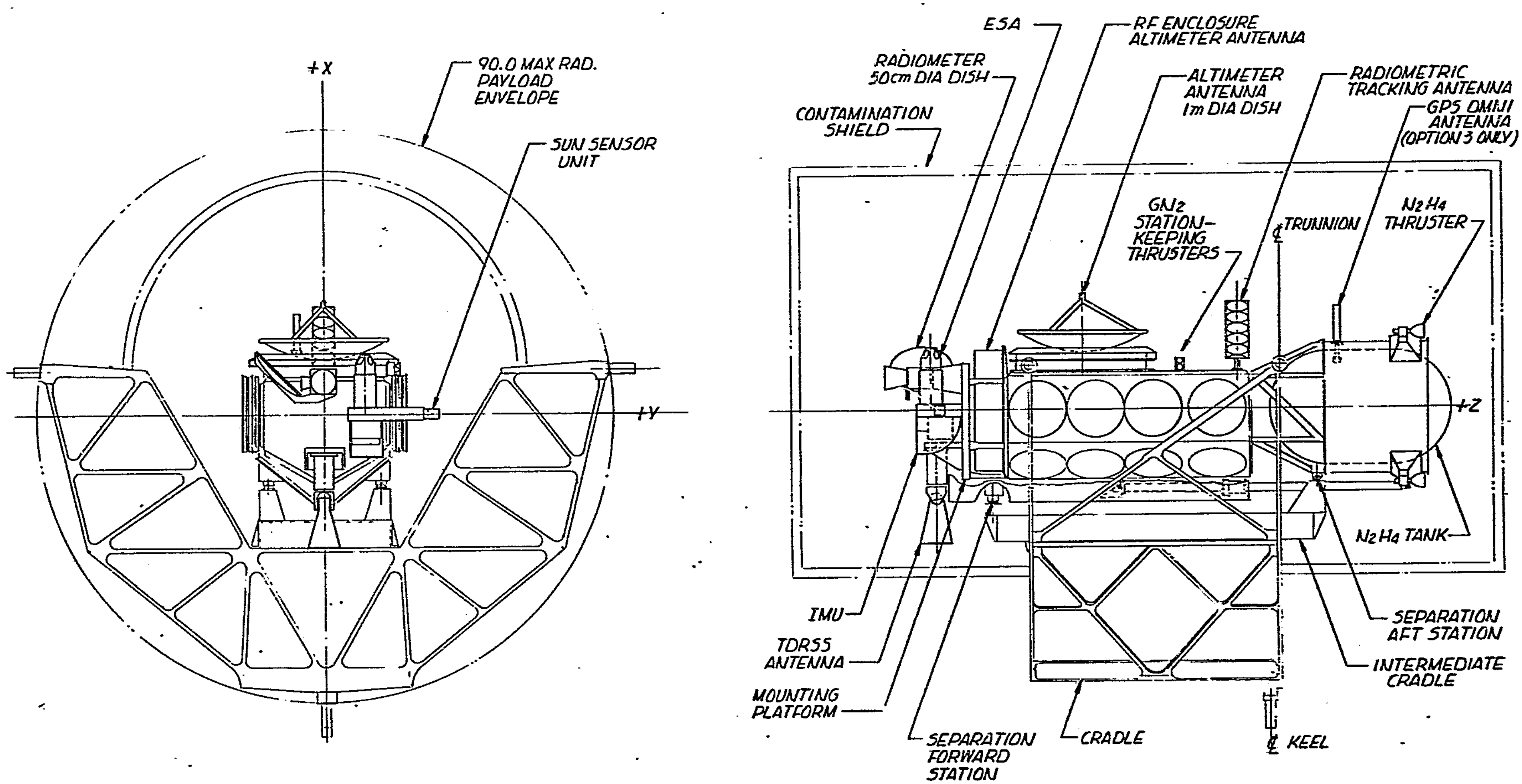

Figure 3-6. TOPEX Options 2 and 3, STS, Launch Configuration

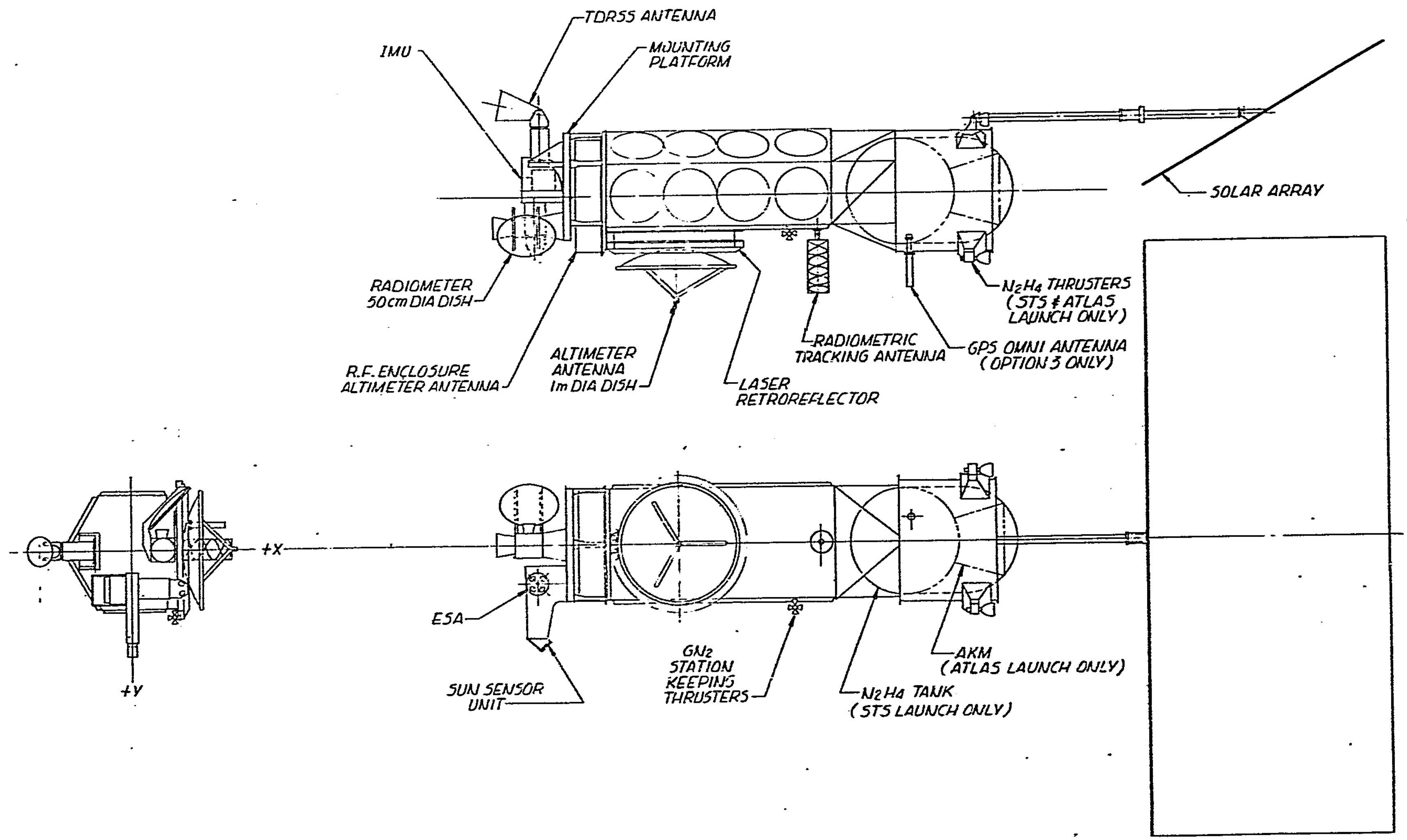

Figure 3-7. TOPEX Options 2 and 3, Orbit Configuration

The 1-meter altimeter antenna will be nondeployable, fixed mounted on the earth-facing panel, and a toroidal laser retroreflector will surround the altimeter dish. A structural bracket on the forward end panel of the ESM will support the earth sensor assembly and sun sensor unit. Also on the forward end panel will be the gimballed TDRS antenna and a fixed, nondeployable radiometer with its half-meter dish.

After orbit injection, the Option 2 and 3 satellites require only that the array deploy (as described for Option 1) and the TDRS antenna be uncaged to transform the satellite to its on-orbit configuration.

SECTION 4.0

CANDIDATE EQUIPMENT

SECTION 4.0
CANDIDATE EQUIPMENT

4.1 STRUCTURE AND THERMAL

4.1.1 STRUCTURE

The TOPEX configurations, as described in Section 3.0, utilize the flight-proven TIROS/DMSP structural concept. The structure consists of four major assemblies: the equipment support module (ESM), the truss, the reaction control support structure (RSS), and the solar array assembly (see Figure 4-1).

The primary function of the ESM is to house the majority of the electronic support equipment. It is pentagonal in section but unsymmetric to provide a large earth-viewing face upon which the instruments (altimeter, radiometer, laser-retro, TRANET-II, and GPS) are mounted. The ESM is approximately 77 inches high and 44 inches across the points of the forming hexagon. The seven ESM panels are constructed of aluminum honeycomb sandwiches with integral machined edge members. The equipment panels are mounted to a lightweight magnesium frame. This construction technique yields a high electronic-to-structure weight ratio while providing a relatively stiff structure (first lateral mode ~8 Hz) which decouples the spacecraft from launch vehicle low frequency inputs. The internal electronic boxes are arranged on all seven panels to achieve an acceptable system in terms of minimizing the mass-balancing weight requirement, maximizing the efficiency of the thermal control subsystem, and functionally grouping the components to minimize harness lengths.

The second major structural component, the RSS, provides mounting locations for components of the propulsion subsystem and those components of the power system which require views to space. The RSS is a 39-inch diameter, 32.5-inch high, riveted aluminum sheet metal cylindrical section reinforced by 18 longerons and 4 external rings. The top ring of the RSS provides the interface to the truss and, for a Space Transportation System (STS) launch, the support for the two aft mounting pads. The intermediate rings provide mounting points for the heavy power and propulsion components which are located on the RSS. The lower support ring forms the spacecraft portion of the interface to the expendable launch vehicle (ELV).

The structural connection between the bottom of the ESM and the top of the RSS is an 11-element titanium weldment truss. The 22-inch long truss provides a high thermal resistance path and thus a low conductive coupling between the RSS and ESM. The elements of the truss are sized to minimize the torsional dynamic coupling between the symmetric RSS and the nonsymmetric ESM.

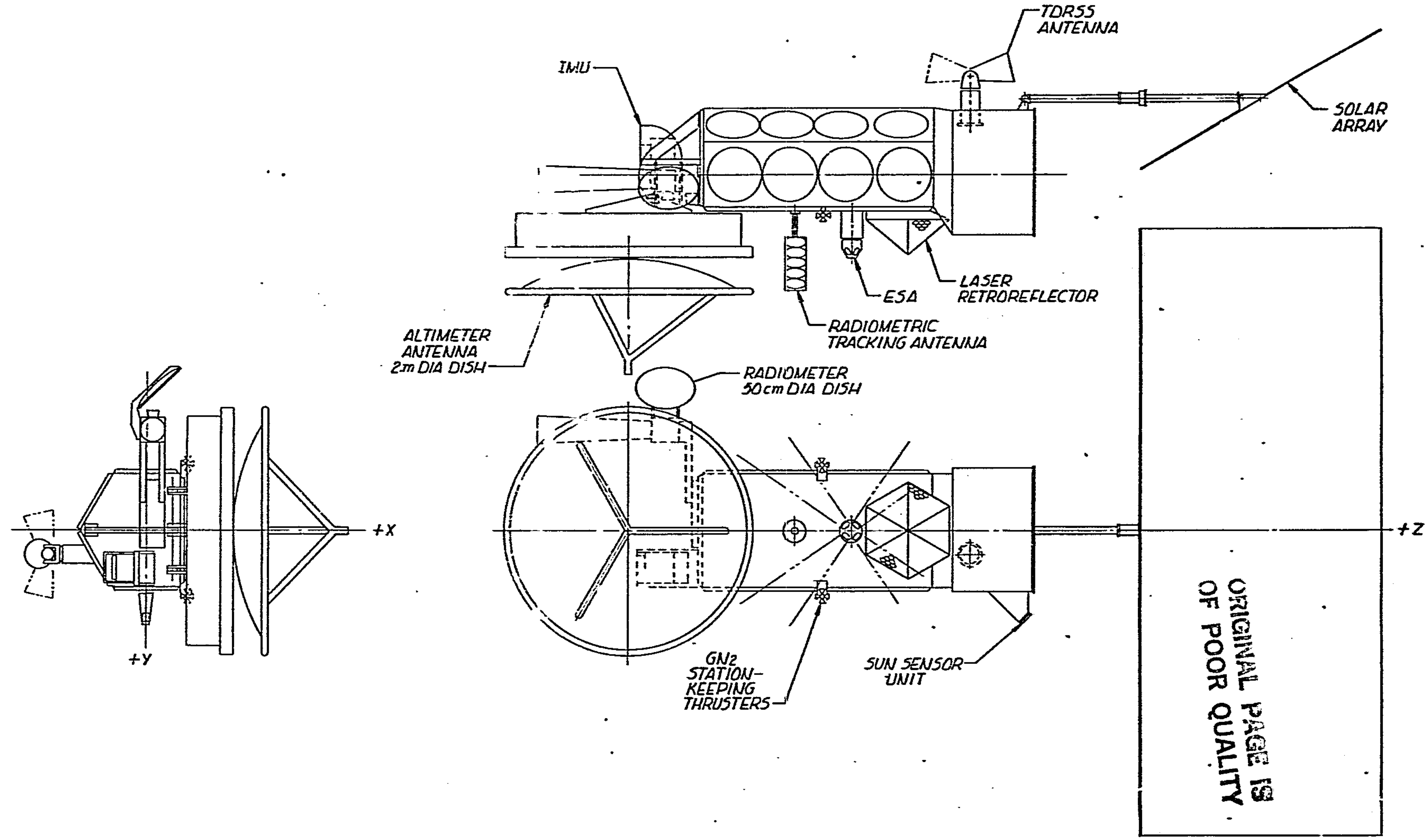

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Figure 4-1. TOPEX Spacecraft

For the Delta launch vehicle configuration, the truss has been eliminated and a short transition section added. Fairing length constraints made it necessary to reduce the overall spacecraft length.

The last major flight structural segment, the solar array system, is composed of the solar array and the solar array support. The solar array consists of eight separate but identical solar panels joined together along their long edges by the deployment assembly hinges. The hinge assembly contains redundant deployment springs, a liquid rate damper, and a position telemetry sensor. Each solar panel substrate is constructed of a honeycomb panel approximately 24 inches wide by 94 inches long, bonded to a riveted sheet metal frame for reinforcement. In the launch configuration, the solar panels are "wrapped" around the four anti-earth sides of the ESM and held by two circumferential cables. The launch loads are transmitted back to the ESM structure through two sets of shear ties located on each of the four ESM panels. In the orbital configuration the solar array is positioned off the +Z end of the satellite and supported by the SAS (shown in Figure 4-2) hinged from the RSS. The long and short booms are tubular aluminum structures which interconnect the mast, the center portion of the solar array, the solar array drive (the device which rotates the array), and the RSS.

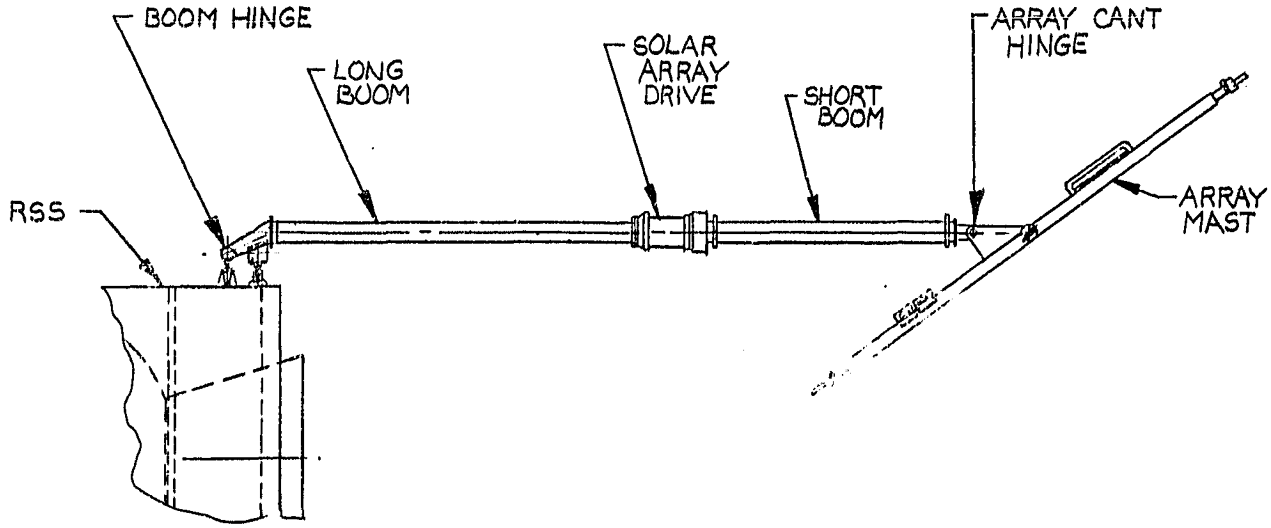

Figure 4-2. Solar Array Support

The spacecraft interface to the launch vehicle is fundamentally different for an ELV and the STS. For an ELV launch, the spacecraft is supported as a cantilever from the +Z end of the RSS. A conical monocoque adapter, which remains as part of the ELV, is utilized to adapt the RSS diameter to the booster interface diameter. The RSS is attached to the adapter during launch with a 20° half angle, two section V-band. There is a bolt cutter on each of the two V-band connecting bolts. The severing of either bolt will release the band and allow spacecraft-booster separation to occur. The actual separation is accomplished by firing the four spacecraft hydrazine engines.

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The Mechanical Airborne Support Equipment (MASE) for an STS launch includes a cradle to support the satellite in the Orbiter, an intermediate structure to interface the satellite and the cradle, and a cocoon to provide contamination and thermal protection for the satellite in the STS. The proposed cradle, an adaptation of the PAM cradle presently under development by MDAC, supports the satellite and intermediate structure horizontally (the satellite Z-axis parallel with the Orbiter X-axis) in the Orbiter. The satellite is supported on the intermediate structure at three points (one at the top of the ESM and two at the top of the RSS) while the cradle interfaces with the STS at five points (four longeron supports, two on each side, and one aft keel fitting) as shown in Figure 4-3. The cradle is constructed of planar truss elements machined from solid aluminum frame with shear skins riveted to the top and bottom, which attaches to the cradle at eight points (four forward and four aft in line with the trunion locations). The intermediate-structure concept permits flexibility in designing the satellite attachment details without necessitating cradle-design modifications. The final element of the MASE, the cocoon, has a fixed lower portion and a 6-element upper portion (three per side) constructed of aluminum truss sections covered by thermal blankets. The upper two of the three sections on each side can be opened by crew command to permit deployment of the TOPEX satellite from the STS. The movable sections are driven by two dc motor/gear train assemblies which also permit the cocoon to be reclosed and latched by crew command. The proposed cocoon design is derived from the PAM thermal shield which is currently under development by MDAC.

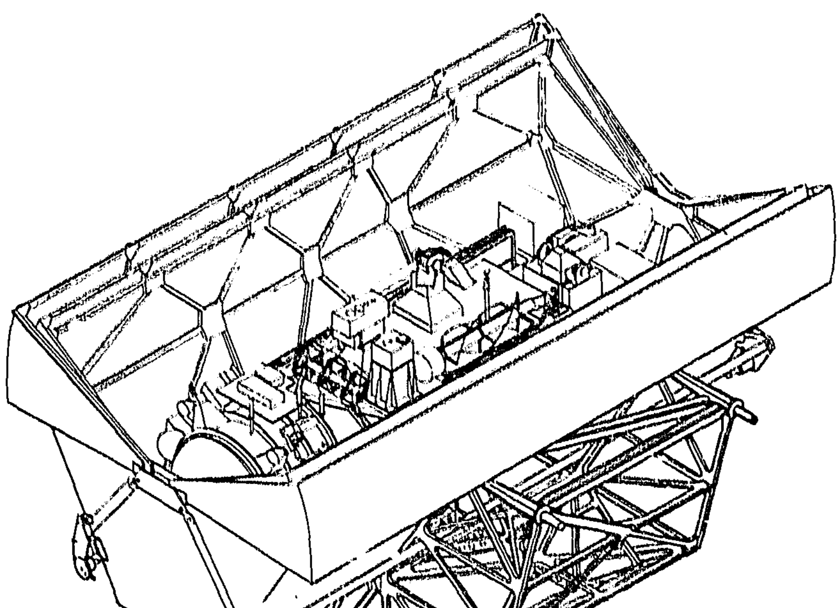


Figure 4-3. Spacecraft In Shuttle MASE

4.1.2 THERMAL CONTROL

Thermal control of the satellite and of the instruments (to the extent required) is accomplished by judicious selection and use of passive and/or active thermal control elements. These elements consist of (1) thermal blankets, (2) selected absorptivity and emissivity finishes, (3) insulators, (4) satellite and equipment layout, (5) heaters, (6) sunshields, (7) louvers with associated thermal control electronics (TCE) units, and (8) thermostats. The layout and finishes are configured to prevent reflected solar energy (glint) from entering the payload fields-of-view. The instruments approach thermal independence in design and in the thermal control methods employed with as small a thermal coupling between them and the spacecraft as practical.

The ESM is thermally controlled by means of multilayer blanketing and 15 TCE driven active pinwheel louvers to maintain component temperatures from +5° to +30°C over all sun angle and dissipation conditions. When required, TCE controlled heaters mounted on each of the ESM panels are used to supply supplemental power to maintain the internal temperatures.

The RSS structure is finished externally with 5-mil aluminized Teflon tape, which has a low solar absorptance-to-emissivity ratio. This causes the structure to run cold during flight. The interior surface is partially covered with aluminized Kapton (aluminum exposed) in the region of the N_2H_4 lines in order to minimize the thermal coupling between that area and the internal hydrazine tank.

All flight-operational electrical components mounted on the RSS (battery, battery charge assembly, array drive, and array drive electronics) are thermally isolated from the structure and have independent thermal control.

The battery packs are temperature controlled by TCE-controlled heaters and rectangular louvers. The -Z surface of the battery packs is painted white. The +Z surface has a TCE-controlled heater attached to prevent the battery temperature from going lower than +5°C during testing or unusual flight conditions.

The battery charge assembly (BCX) is thermally decoupled from the RSS via polycarbofil isolators and thermal blankets, and is passively controlled by a fixed radiator and by its location. The radiator prevents excessively high temperatures during peak battery charging, and the orientation (earth facing) of the BCX prevents excessively cold temperatures. The radiator is finished with 10-mil aluminized Teflon to minimize solar inputs.

The array drive electronics (ADE) is controlled by use of thermal blankets, a fixed radiator area on the anti-satellite surface covered by 2-mil aluminized Teflon tape, and a TCE controlled heater.

The N_2 thrusters (eight units), pressure regulator, and latch valve are thermally isolated from the RSS, and are blanketed to minimize heat loss and external inputs. Redundant, thermostatically-controlled heaters are provided on each element (total of 20 thermostat heater sets) to prevent temperatures from falling below 5°C.

The entire hydrazine system is heated to maintain a bulk temperature above 10°C until the completion of the ascent phase. At that time, the heater system is deactivated and the hydrazine allowed to freeze.

4.2 ATTITUDE DETERMINATION AND CONTROL

4.2.1 LAUNCH VEHICLE OPTIONS

The ATN baseline used in this study for application to TOPEX was designed and demonstrated on an Atlas launch vehicle. The Attitude Determination and Control Subsystem (ADACS) provides not only mission mode precision pointing but also provides navigation and guidance through the boost phase of the mission. As a result, it was directly applicable to the STS portion of the SAATN study where guidance and control was required during propulsion maneuvers after separating from the Shuttle. This SAATN effort was used extensively in the TOPEX study. In fact, because the Delta option requires mission mode pointing control only, the ATN system is directly applicable, with potential on-board software and hardware reduction. (The hardware reductions are primarily in the propulsion system, with only the accelerometers in ADACS not required on Delta.) Since the ATN is designed for Atlas, and the STS option for TOPEX has additional control requirements over the Delta option, we will emphasize the STS system in the remaining paragraphs.

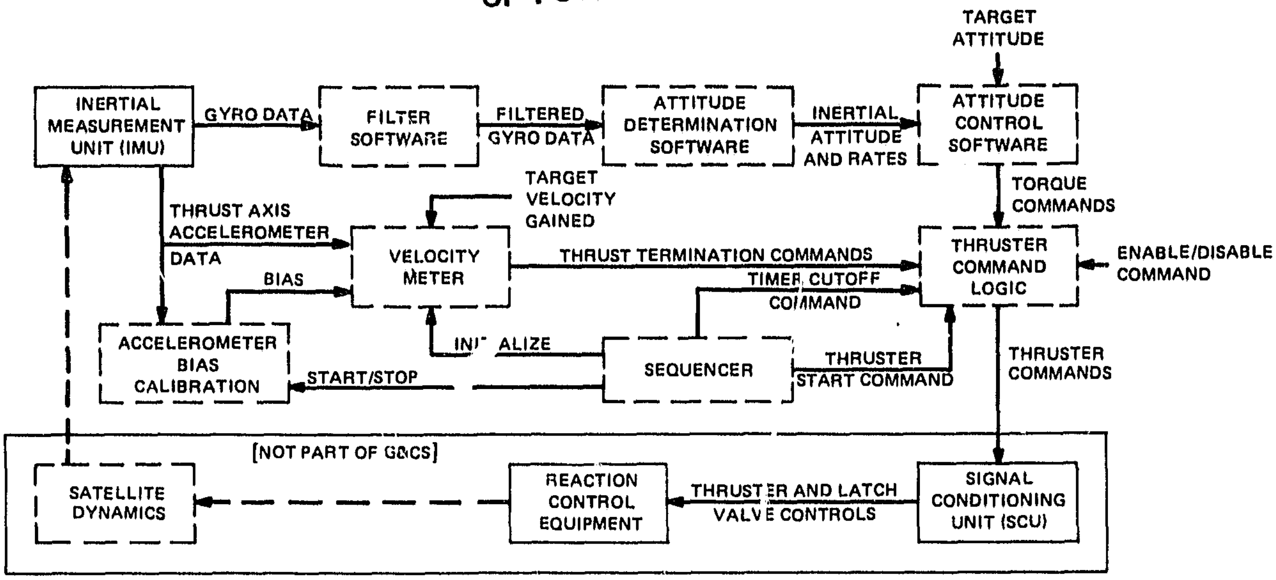
4.2.2 ADACS DESCRIPTION

The ADACS functionally integrates the on-orbit attitude control hardware, the ascent guidance hardware, and the requisite software to provide, in conjunction with the Reaction Control System, all ascent guidance and control functions.

For STS launches, TOPEX will utilize an all-liquid propellant configuration which incorporates a large hydrazine propellant tank, developed for the Viking program, feeding four 100-lb thrust hydrazine engines. The guidance system uses the thrust axis velocity meter concept which integrates data from a thrust axis accelerometer exlusively. A computer generated timing signal is provided to limit burn duration as a backup to the velocity meter. A functional block diagram of the configuration for STS mission ascent maneuvers is shown in Figure 4-4.

Throughout the burns, inertial attitude is maintained by utilizing a closed-loop, gyro-sensed control which is referenced to a target attitude uploaded from the ground. Initialization of the inertial attitude reference is accomplished prior to the perigee kick burn of the two-burn transfer sequence by a yaw gyrocompassing attitude determination algorithm, in which earth sensor and gyro data are combined in a simple Kalman filter.

The STS mission ascent control system for TOPEX will be identical to that designed for TIROS-N/ATN. During all hydrazine burns, motion about the thrust axis is controlled by GN_2 thrusting, and motion about the remaining two control axes is provided by off-pulsing the hydrazine engines. During the coast and reorientation segments of each ascent sequence, as well as during array (boom) and antenna deployments, all three axes are controlled by GN_2 thrusting.



NOTE: ALL HARDWARE/CPU INTERFACES ARE VIA THE CIU

Figure 4-4. ADACS Functional Block Diagram for STS Mission Ascent Maneuvers

For Atlas-F launches, both three-axis gyro and three-axis accelerometer data are utilized. The ADACS provides navigation from liftoff until separation from the Atlas booster, and subsequently will navigate, guide, and control the vehicle to orbit injection. This is the same approach that has been utilized for all TIROS-N type missions and will be utilized for all ATN launches.

For Delta launches, there are no guidance requirements on the TOPEX spacecraft because they are provided by the booster.

The mission orbit attitude control system will be essentially the same as that utilized for ATN. However, to minimize program cost on the STS launch option, conical scanning earth sensors, which are required for attitude initialization on STS-launched missions, will also be utilized during the mission orbit, thereby replacing the TIROS-N/ATN static ESA. Use of the conical sensors results in achievable pitch and roll attitude determination accuracy of 0.15 degree, 3σ. Some changes in software mission constants from those of ATN are also probable to accommodate altered satellite mass properties. A functional block diagram of the ADACS Orbit Mode Control Subsystem is shown in Figure 4-5. A summary of the ADACS equipment complement to be carried on TOPEX is given in Table 4-1.

4.2.3 ASCENT ATTITUDE CONTROL

The ascent attitude control system for STS launches is functionally the same as the hydrazine trim burn attitude control system design employed on TIROS-N/ATN. (A typical Atlas ascent sequence is shown in Figure 4-6.) The system must (1) orient the thrust axis for the orbit adjust burns, (2) maintain the orientation during the burns, and (3) provide coarse inertial pointing at other times. During the hydrazine burns, motion about the thrust axis is controlled by the GN_2 thrusters, and motion about the two control axes transverse to the

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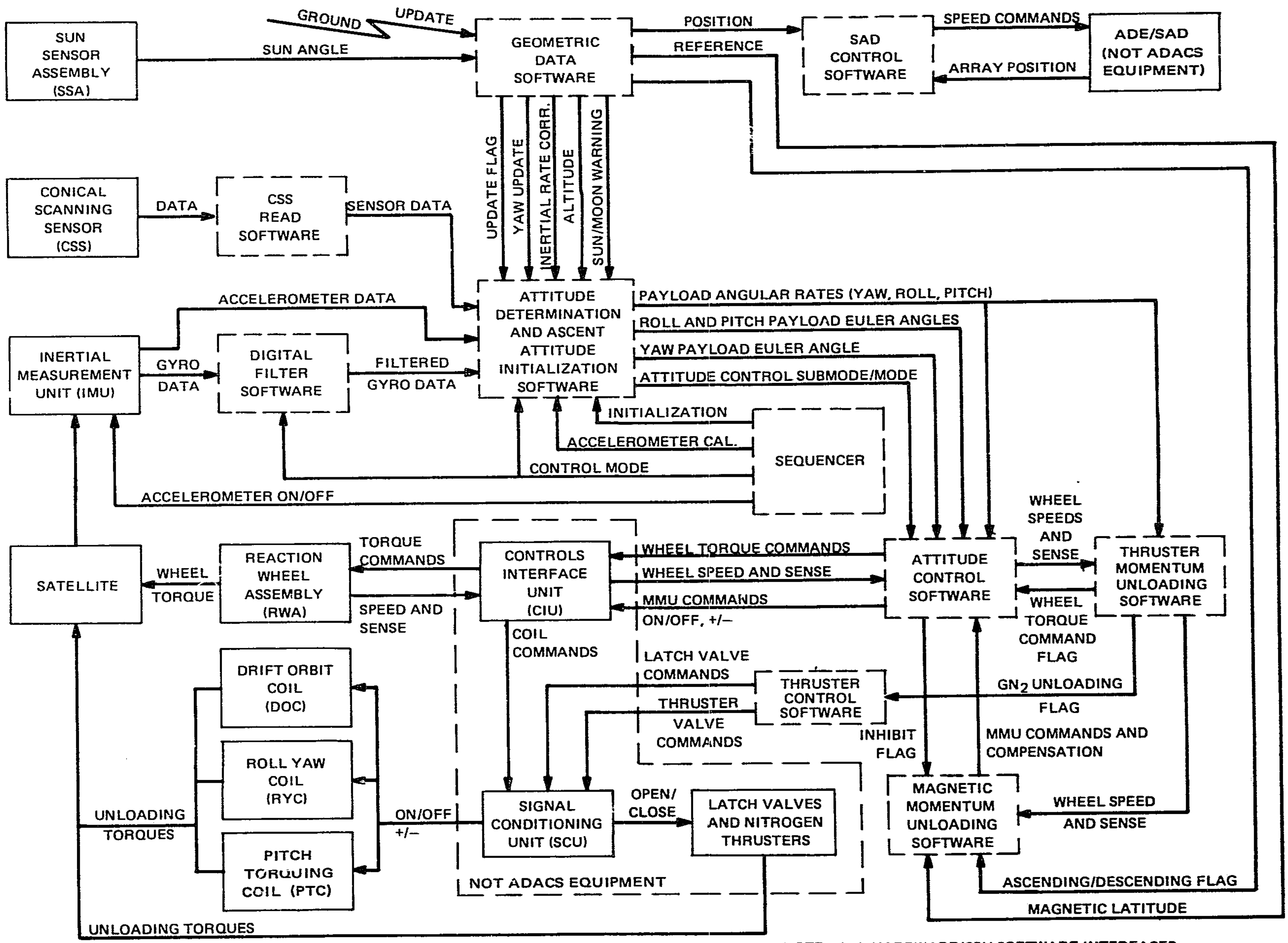
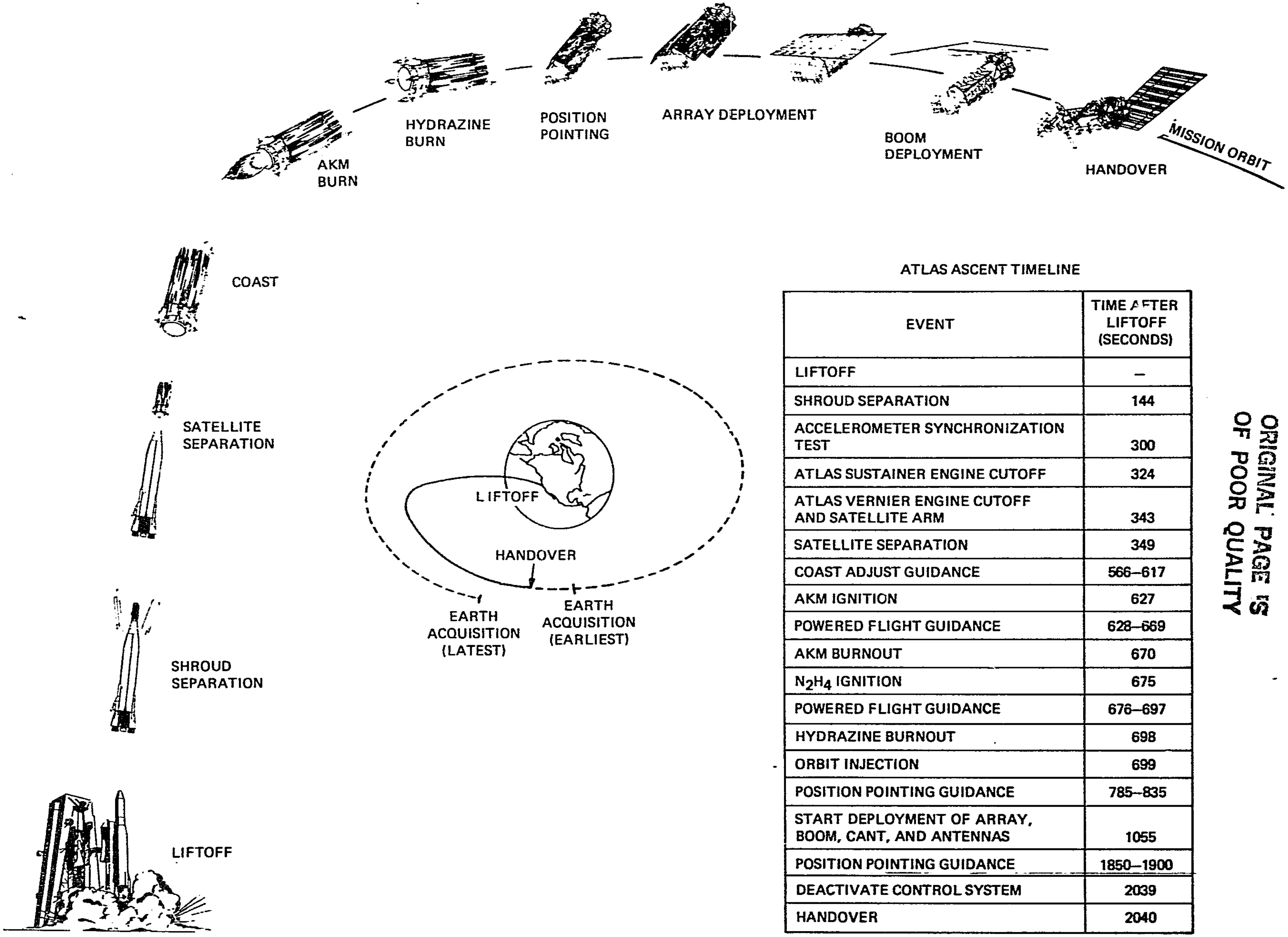


NOTE: ALL HARDWARE/CPU SOFTWARE INTERFACES ARE VIA THE CIU.

Figure 4-5. Orbit (Drift and Mission) Mode Control Subsystem Configuration Block Diagram



ATLAS ASCENT TIMELINE

EVENT	TIME AFTER LIFTOFF (SECONDS)
LIFTOFF	–
SHROUD SEPARATION	144
ACCELEROMETER SYNCHRONIZATION TEST	300
ATLAS SUSTAINER ENGINE CUTOFF	324
ATLAS VERNIER ENGINE CUTOFF AND SATELLITE ARM	343
SATELLITE SEPARATION	349
COAST ADJUST GUIDANCE	566–617
AKM IGNITION	627
POWERED FLIGHT GUIDANCE	628–669
AKM BURNOUT	670
N_2H_4 IGNITION	675
POWERED FLIGHT GUIDANCE	676–697
HYDRAZINE BURNOUT	698
ORBIT INJECTION	699
POSITION POINTING GUIDANCE	785–835
START DEPLOYMENT OF ARRAY, BOOM, CANT, AND ANTENNAS	1055
POSITION POINTING GUIDANCE	1850–1900
DEACTIVATE CONTROL SYSTEM	2039
HANDOVER	2040

Figure 4-6. Typical Atlas/SAATN Guidance Mode Sequence

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TABLE 4-1. TOPEX ADACS EQUIPMENT COMPLEMENT

Equipment	Supplier
Conical Scanning Sensor Assembly (CSS) - STS	Barnes or Ithaco
Earth Sensor Assembly (ESA) - Delta/Atlas	Barnes
Inertial Measurement Unit (IMU)	Honeywell
Reaction Wheel Assembly (RWA)	RCA AE/CSDL
Precision Sun Sensor (SSA)	Adcole
Magnetic Torquers:	
Roll/Yaw Torquing Coil (RYC)	RCA AE
Torquing Coil	RCA AE

thrust axis will be controlled by off-pulsing the hydrazine engines. During coast and reorientation, all axes are controlled by the GN_2 thrusters. The system design goals, based upon the results of the guidance system analysis, are to provide control to within 1.0 degree about the thrust axis and to within 0.2 degree about the transverse axes.

To meet these requirements, the bang-bang control system, functionally illustrated in Figure 4-7, is used.

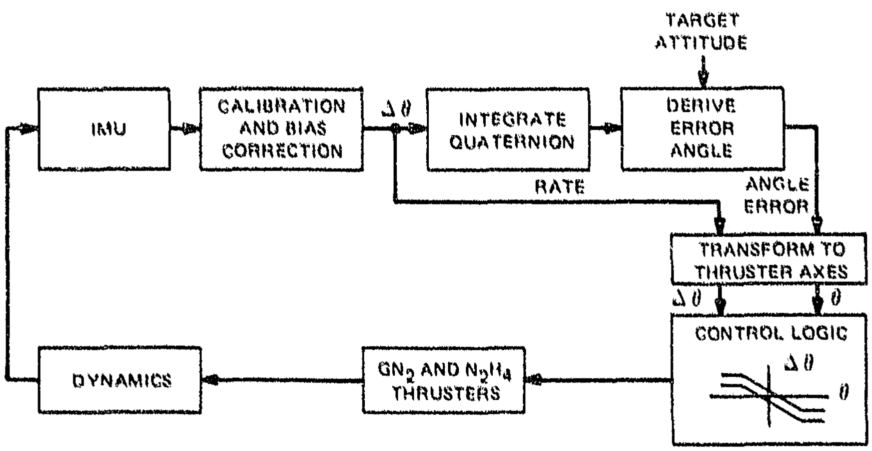

Figure 4-7. Ascent Control System Functional Block Diagram

4.2.4 MISSION ORBIT ATTITUDE DETERMINATION AND CONTROL

The mission-orbit Attitude Determination and Control Subsystem is functionally the same for all booster options as that developed for TIROS-N/ATN. Specifically, via the interaction of software elements and hardware as depicted in Figure 4-5, the subsystem performs the following functions:

(a) Provides for automatic closed loop acquisition of the geodetic local vertical and the orbit normal after "handover" from the Ascent Guidance Phase Control Subsystem.

(b) Provides for automatic closed-loop reacquisition of geodetic local vertical and orbit normal.

(c) Provides for on-board three-axis satellite attitude determination.

(d) Provides for maintenance of proper orientation of the satellite body fixed-coordinate reference with respect to the reference geoid by active, closed-loop, three-axis control.

(e) Provides ground stations with earth sensor, sun sensor, and gyro telemetry data for three-axis satellite attitude determination.

(f) Provides for automatic removal of accumulated momentum on the satellite by magnetic torquing as a primary means and by GN_2 thruster torquing as a backup means.

(g) Provides required signals for telemetry status and monitoring parameters pertaining to subsystem operation.

(h) Provides attitude control during stationkeeping ΔV maneuvers.

The subsystem uses Reaction Wheel Assemblies (RWA's) as the control element with magnetic torquing backed up by GN_2 thrusting for momentum management. Four RWA's are used on each satellite, one on each of the three orthogonal control axes and a fourth for redundancy. The fourth RWA is mounted in a skewed orientation so as to be equiangular with each of the three control axes. The system attitude reference is derived either from two conical scanning earth sensors (CSS's) on STS or earth sensor assemblies (ESA's) on Delta and Atlas for pitch and roll and inertial data with once-per-orbit sun sensor (SSA) updates for yaw. Several of the TOPEX satellite options in their mission configurations were shown in Section 3.0. In the mission orbit operation, the X-axis is the yaw axis, the Y-axis is the roll axis, and the Z-axis is the pitch axis. The satellite X-axis is maintained to within 0.2 degree (3σ) of the local geodetic reference in the presence of both internal and external disturbances. Error rates about the pitch and yaw axes are maintained at less than ± 0.035 degree/second and about the roll axis at less than ± 0.015 degree/second.

The mission orbit mode control subsystem begins functioning at handover, following full deployment of the solar array and all antennas. It has five operating submodes whose functions are as follows:

- Rate Nulling (RN) - This is basically an inertial hold mode wherein the reaction wheel torque commands for the yaw, roll, and pitch control loops consist of terms that are proportional to the gyro-derived rate compensated for bias effects and the integral of gyro rate.

- Search (SCH) - The Search submode control laws are basically the same as those for the Rate Nulling except for the insertion of selectable command rates into the pitch and roll loops. The applied command rates are a function of the capture state of the two CSS's (or ESA's); i.e., which of the CSS's is viewing the earth and where the earth lies in each CSS scan.

- Yaw Gyrocompassing (YGC) - In the YGC submode, the roll and pitch loops operate under CSS (or ESA) control. The control laws for these axes contain terms proportional to CSS- (or ESA-) derived attitude error, gyro-derived attitude error rate, and the integral of attitude error. Gyro-derived roll and yaw rates furnish the required information for yaw axis control. The gyro-derived roll rate furnishes information proportional to yaw attitude error, measuring a component of orbit rate proportional to yaw; the gyro-derived yaw rate supplies the damping term in the control law.

- Nominal (NOM) - The pitch and roll control laws in this case are the same as those in YGC, employing CSS (or ESA) attitude measurements, attitude error rate, and the integral of attitude error. In the yaw loop, this same philosophy is also utilized, with yaw attitude being derived from an integration of yaw-axis rate data referenced to once per orbit sun sensor updates.

- Coast - The Coast control submode is entered when there is a temporary loss of CSS (or ESA) data while operating in the Nominal mode. In Coast, the pitch and roll control laws are the same as those in Rate Nulling, and the yaw control law is the same as that in Nominal.

Magnetic dumping on the pitch axis is provided by magnetic torquing from the pitch torquing coil (PTC) and on the roll and yaw axes by the roll/yaw coil (RYC).

Whenever the component of system momentum along one of the body reference axes exceeds a specified limit, the RCS backup GN_2 desaturation system will automatically reduce momentum. For this system, which is the same as that designed for TIROS-N/ATN, the momentum calculation utilizes gyro-sensed body rates as well as measured RWA rates. When momentum is excessive, thrusters are fired for 0.1 second to produce a desaturating torque about the appropriate axis.

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The geometry between the sun vector and the orbit plane changes continuously in the TOPEX orbit. This is not the case on TIROS or DMSP, because they utilize a sun synchronous orbit. As the sun passes through the orbit plane, it is necessary to reorient the spacecraft to assure adequate power and thermal conditions. This flip maneuver is required on DMSP and is accomplished by using the yaw RWA to reorient the pitch axis (Z-axis) from the positive orbit normal to the negative orbit normal (and vice-versa).

During the infrequent stationkeeping maneuvers with the cold gas nitrogen thrusters, the torque and momentum disturbance levels are sufficiently low that they can be handled by the RWA's. Performance improvements are possible after in-flight calibration.

4.3 DATA HANDLING AND COMMAND

4.3.1 COMMAND

A block diagram of the command subsystem is shown in Figure 4-8, and the characteristics are described in the following paragraphs.

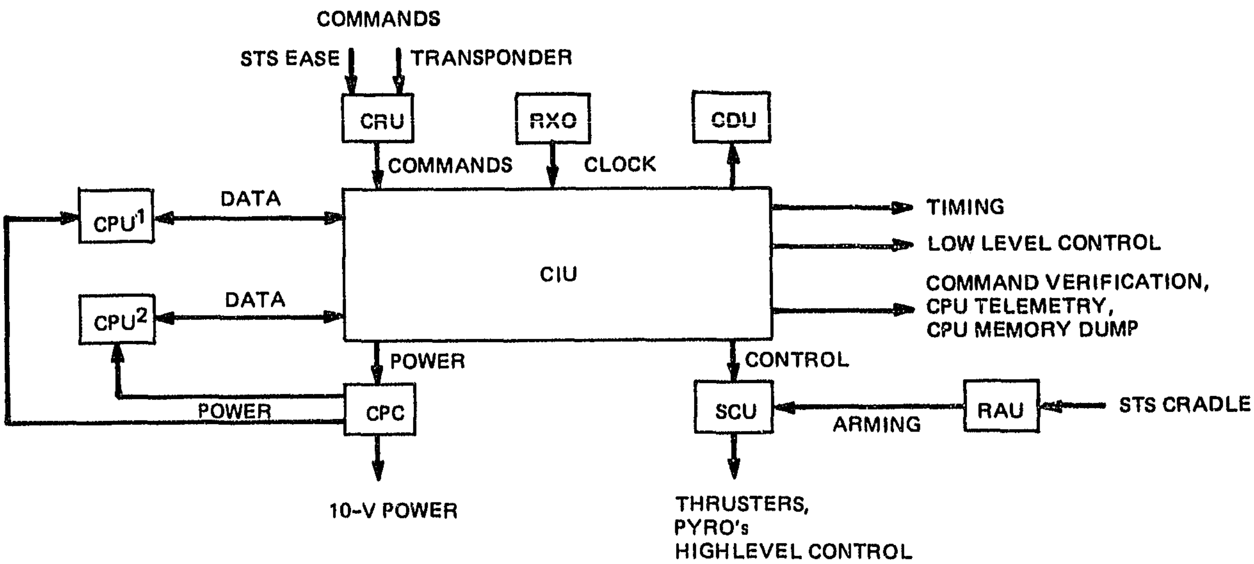

Figure 4-8. Command Subsystem Block Diagram

4.3.1.1 Controls Power Converter (CPC)

This unit, although physically located within the CIU, is generally considered a part of the Power Subsystem. It provides 10-volt power to the CIU and CPU's and +10-volt power to spacecraft bus units and instruments for use in powering interface circuits.

4.3.1.2 Central Processing Units (CPU's)

The computers are redundant in that either CPU can perform the entire mission. The CPU's will be based on the DMSP design, with each containing a 32K read/

write memory with single-error-correction/multiple-error-detection capability for fault tolerant processing. The computers are loaded prior to liftoff in either the STS or expendable launch vehicle configuration and can be reloaded in-orbit if reprogramming is desired. Memory size is such that the initial load contains all required codes for STS ascent, STS parking orbit, ascent to final orbit, and final orbit operation.

4.3.1.3 Redundant Crystal Oscillator (RXO)

The RXO will be a dual, oven-controlled crystal oscillator which meets the TOPEX stability requirement of 1 part in 10^{10} per day. It will be derived from equipment used on the RCA NOVA program. (The TIROS RXO is specified at 1 part in 10^8/day and must be replaced for TOPEX.) The RXO provides the timing for the Command and Data Handling Subsystems, as well as for the TOPEX instruments. The spacecraft bus utilizes binary divisions of a 5.12-MHz source to provide timing in multiples and fractions of seconds. For this reason, the Command and Data Handling Subsystem components require 5.12 MHz, not 5 MHz. If the TOPEX instruments require precisely 5 MHz, this will be provided through frequency synthesis in the cross strap unit (see Section 4.3.2) in a manner similar to that currently used on TIROS spacecraft to provide data recording and playback clocks. Except for phase-locked-loop jitter, these clocks have the same stability as the RXO, from which the input to the synthesizer is obtained.

4.3.1.4 Signal Conditioning Unit (SCU)

The SCU provides high level control signals for activating thrusters, pyrotechnics, propulsion latch valves, heaters, and attitude control magnetic torquing coils. For the TOPEX mission, the SCU will also perform the Remote Arming Unit (RAU) functions described below, as was proposed for the SAATN program. In addition, current TOPEX instrument command requirements are for switched relay contacts and for relay drivers. If this requirement remains, a CDU will be added to include the necessary relays and drivers, activated by low level control signals from the CIU.

4.3.1.5 Remote Arming Unit (RAU)

The RAU is basically a timer which provides the necessary inhibits to meet STS safety requirements. An external programming plug inhibits the RAU functions for a non-STS launch. Independent control of the SCU, pyro, and hydrazine functions is possible from the STS electrical airborne support equipment (EASE) and from ground test equipment. In normal operation, the SCU and pyro functions will be enabled by the RAU when the spacecraft separation from the STS exceeds the distance required for safety. Similarly, at a second distance, the hydrazine system will be enabled.

4.3.1.6 Command Reformatting Unit (CRU)

The TIROS/DMSP controls interface unit (described below) is built to accept commands demodulated from a ternary bit stream from a Command and Data Acquisition (CDA) station. To be compatible with the binary STDN/TDRS command stream, the CRU is used to transform the demodulated STDN/TDRS signal to the three lines ("1", "0", "S") used by the CIU. The CRU will be that proposed for SAATN. In addition to the reformatting function, the CRU will provide a selection of uplink sources between the onboard transponder and the STS EASE.

4.3.1.7 Controls Interface Unit (CIU)

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The CIU provides the interface between the CPU's and the rest of the satellite, distributing data, low level control signals, and timing and inputting data for processing by the CPU's. The present CIU will be modified as proposed for SAATN satellites to function as required during phases of the mission prior to the final mission orbit. The present CIU has over 120 low-level discrete control bits available for use directly by instruments, or indirectly, through SCU and CDU relays and drivers. The total number of commands required by TOPEX instruments is not known but presumably, will not exceed 120. If it does, an annex to the CIU (an existing design) provides 96 bilevel control lines and 96 hardware-controlled-pulse-duration pulse control lines. This annex is not proposed for TOPEX. CIU pulse commands have a duration which is under software control and can range from one half to one and one-half seconds. If TOPEX instruments are designed to accept logic level command signals at 1 $\pm$0.5 second pulse duration, the TIROS CIU will require no change for commands. If a 100-millisecond relay drive or relay contact closure is required for TOPEX, the CIU and SCU changes described above will be made.

4.3.2 DATA HANDLING

A block diagram of the Data Handling Subsystem is shown in Figure 4-9, and the characteristics are described in the following paragraphs.

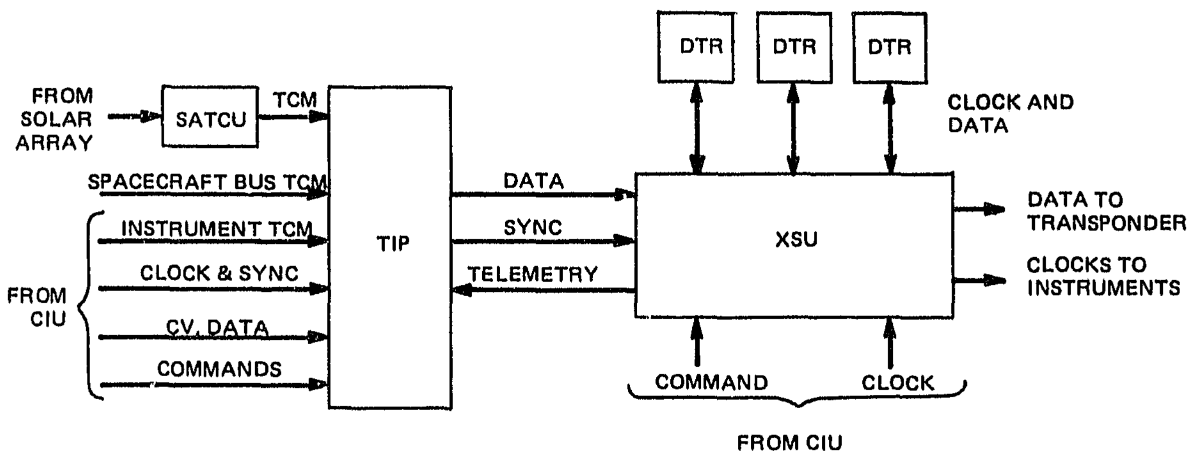

Figure 4-9. Data Handling Subsystem

4.3.2.1 Solar Array Telemetry Commutation Unit (SATCU)

This unit commutates 16 telemetry points from the solar array (hinge angles, temperature, and voltage) into a single data stream for insertion into the TIP data stream.

4.3.2.2 Digital Tape Recorder (DTR)

The tape recorders allow instrument and spacecraft bus data to be recorded and played back at optimum times. The TIROS DTR's each consist of two independent transports and a common electronics section. These DTR's record in one of two

modes. In the low data rate mode, one track is used for a capacity of 1.125×10^8 bits per transport (2.25×10^8 bits per DTR). A second mode is used for high rate recording. In this mode, four tracks are used, for a total of 4.5×10^8 bits per transport. For the TOPEX mission, three recorders will be carried, two for routing use and one as a backup. This will allow one recorder to play back as the second is recording. In the low rate record mode, a capacity of 4.5×10^8 bits plus 2.25×10^8 bits spare are available. This exceeds the TOPEX requirement for 3.6×10^8 bits. The record and playback rates are determined by clocks supplied by the cross strap unit clock synthesizer. TOPEX data will be recorded at 16 kbps and played back at either 48 kbps or 500 kbps. At this recording rate, each transport can record one orbit of TOPEX data. This data will then be played back through TDRS or directly to the ground at approximately a 3:1 playback rate through the Q channel in the multiple access (MA) mode, or at approximately a 30:1 playback rate through the Q channel in the single access (SA) mode. Real-time transmission of the science data will be at 16 kbps, and housekeeping data will be at 2 kbps.

4.3.2.3 Cross Strap Unit (XSU)

The XSU will provide the required clock and sync signals to the instruments. As mentioned in Section 4.3.1, if the instruments need 5 MHz rather than the 5.12-MHz used by the spacecraft bus, the 5 MHz will be synthesized in the XSU from an input directly related to the RXO, the spacecraft master oscillator. The XSU provides all data path steering required to record TIP data, playback recorded data, and transmit data in real-time.

4.3.2.4 TIROS (or, in this case, TOPEX) Information Processor (TIP)

The TIP gathers housekeeping telemetry from throughout the satellite, computer telemetry and memory dumps, command verification, and instrument scientific data and formats the data into a 16-kbps data stream which also includes TIP overhead such as time code and frame counter. The data stream will consist of approximately 14.5-kbps scientific data and 2-kbps housekeeping data. The present TIP can operate at either 8.32 kbps or 16.64 kbps, although some changes will be required in the PROM-controlled formatting logic. Also, the present TIP time code is in days (up to 512) and milliseconds of day. This will be changed to meet the TOPEX requirement of an 8-year rollover with increments of microseconds.

A second data output format will be available for use as a beacon. This data will be the 2-kbps described above and will not be available when the 16-kbps mode is in use.

4.4 TELECOMMUNICATIONS

A block diagram of the Telecommunications Subsystem is shown in Figure 4-10, and the characteristics are described in the following paragraphs. The primary elements of the Telecommunications Subsystem will be the TDRS transponder, S-Band transmitter, and the various antennas.

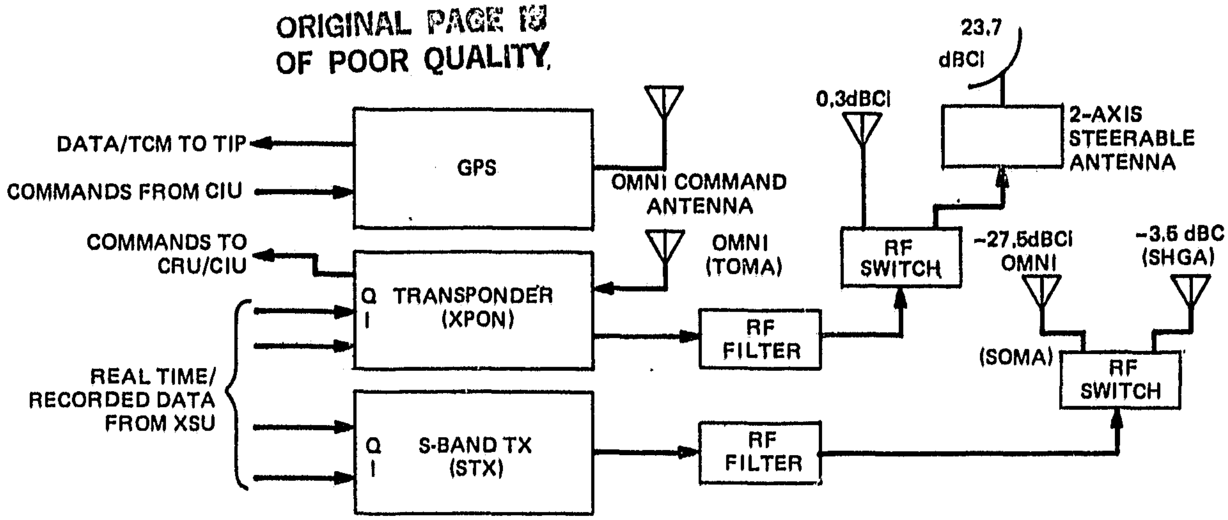

Figure 4-10. Telecommunications Subsystem Block Diagram

The transponder will be similar to the one used on the Dynamics Explorer spacecraft. It will operate at the specified frequencies of 2107.40625 MHz on the forward link and 2287.50000 MHz on the return link, with 5 watts RF output power. Because of the required data rates (see Table 4-2), the high-rate playback will use the TDRS Data Group 1, Mode 3 design. For the other downlink modes of operations, the transponder will operate in TDRS Data Group 1, Mode 2. The independent ground-direct S-Band link will use a TDRS-compatible 5-watt S-Band transmitter, incorporating dual channel (Q and I) operation.

In general, Multiple Access (MA) service on TDRS requires 10 dB more gain than Single Access (SA) service. Each TDRS satellite can accommodate 1 MA user in the forward direction and 20 in the return direction, whereas only 2 SA users in the forward direction and 2 in the return direction can be accommodated. To obtain omni command capacity, the SA forward mode must be used. Table 4-3 is the link calculation for commanding TOPEX via TDRS.

TABLE 4-2. TOPEX DATA RATES

Channel	High Rate (kbps)	Low Rate (kbps)
Uplink		
TDRS Command Channel (SA)	1	N/A
Downlink		
TDRS Q Channel (SA mode)	500	16
I Channel (SA mode)	2	2
Q Channel (MA mode)	N/A	48
I Channel (MA mode)	N/A	2
Q Channel (SA mode)	N/A	2
I Channel (SA mode)	N/A	2
Ground-Direct Q Channel	500	N/A
I Channel	2	N/A

Note: Data rates on the Q channel greater than 150 kbps require use of TDRS Data Group 1, Mode 3.

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TABLE 4-3. SSA FORWARD LINK FROM TDRS 1 KBPS

Parameter	Units	Value	Comments
TDRS EIRP	dBW	43.5	SSA Service
Free Space Loss	dB	-191.6	Loss for 42510 km distance
Polarization Loss	dB	- .6	Assume
Spacecraft Passive Loss	dB	- 1.2	Assume
Spacecraft Antenna Gain	dBi	- 3.0	
Effective Received Power	dBW	-152.9	
Spacecraft Noise Density, N_o	dBW/Hz	-199.3	T_{eff} = 865°K at receiver input
Spacecraft P/N_o	dB-Hz	46.4	
Required for Acquistion	dB-Hz	39.5	1 kbps command rate
Margin-Acquisition	dB	6.9	
Spacecraft P/N_o	dB-Hz	46.4	
Command/Total Power	dB	-0.5	NASA STD TDRS/STDN user transponder
Transponder Loss		-2.4	
Command Channel P/N_o	dB-Hz	43.5	
Data Rate	dB-Hz	30.0	1 kbps
E_b/N_o	dB	13.5	For BER = 10^{-5} transponder requirement
Required E_b/N_o	dB	10.5	
Margin-Data Detection	dB	3.0	

Table 4-4 shows required antenna gains for the various links and mission phases for 5-watt and 10-watt transmitters, for both the 9-foot and 26-foot STDN antennas, and for SA and MA service, with the selected mode indicated. Examples of link calculations with required antenna gain for both TDRS and STDN links are shown in Tables 4-5 and 4-6.

Based on the tables cited, one omni antenna will be used with the transponder to provide 2 kbps beacon data when no other data source is being transmitted. Since the SA service is required to provide the omni coverage, actual use of this link will be on an intermittent basis. For the independent ground-direct S-Band STDN link, one omni antenna will be used for beacon data and one shaped antenna will be used for high rate playback. RF switches will connect the transmitter/transponder to the proper antenna. The primary antenna for TDRS return links will be a medium-high gain, two degree of freedom, steerable dish. This will be used to transmit bursts of real-time science data using MA service, low-rate recorder playback using MA service, and high-rate playback using SA service. In each case, the above data will appear on the Q channel with real-time housekeeping data on the I channel. A summary of the link characteristics is given in Table 4-7.

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TABLE 4-4. REQUIRED ANTENNA GAINS

Link	Station	Spacecraft Atttiude	Pattern	Data Rate	Required Gain for 3-dB Margin	
2107.40025 MHz Command	TDRS	Random	Omni	1 kbps	-3 dBci	
					5 Watts	10 Watts
2287.50000 MHz Beacon	TDRS	Random	Omni	2 kbps	SA -0.29 MA 9.81	-3.39 6.81
2287.50000 MHz Real-Time Science Data	TDRS	Stabilized	Shaped	16 kbps	SA 8.74 MA 18.84	5.74 15.84
2287.50000 MHz Low-Rate Playback	TDRS	Stabilized	Shaped	48 kbps	MA 23.61	20.61
2287.5000 MHz High-Rate Playback	TDRS	Stabilized	Shaped	500 kbps	SA 23.69	20.69
					5-Watt 26-ft Dish	5-Watt 9-ft Dish
High-Rate Playback	STDN	Stabilized	Shaped	500 kbps	-12.6	-3.5
Beacon	STDN	Random	Omni	2 kbps	-36.6	-27.5
High-Rate Play-back STS Orbit	STDN	Stabilized	Shaped	500 kbps	-23	-13.9
Beacon STS Orbit	STDN	Random	Omni	2 kbps	-45	-35.9

A GPS antenna, receiving, and processing system will be used to provide the necessary accuracy for TOPEX ephemeris determination. Control of the medium-high gain TDRS antenna will be open-loop, using CPU determination of the required position (based on ephemeris) and control via the CIU/CDU.

4.5 PROPULSION SUBSYSTEM

4.5.1 LAUNCH VEHICLE OPTIONS

4.5.1.1 STS-Launch Application

For the STS-launched TOPEX configuration, the Propulsion Subsystem is used to transfer the satellite from the STS parking orbit to the final operational orbit. We have based the TOPEX system on the SAATN study which resulted in a

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TABLE 4-5. TYPICAL TDRS RETURN LINK ANALYSES WITH REQUIRED ANTENNA GAIN

Parameter	Units	SA Beacon	MA Low-Rate Playback	SA High-Rate Playback	Remarks
Spacecraft Transmit Power	dBW	7	7	7	5 Watts
Spacecraft Passive Loss	dB	-1.3	-1.3	-1.3	Connectors, switch, mismatch
Spacecraft Antenna Gain	dBci	-0.3	23.6	23.7	
Spacecraft EIRP	dBW	5.4	29.3	29.4	Sum of above
Required EIRP-Acquisition	dBW	-6.0	+4.0	-6.0	SA/MA/SA STDN 101.2
Margin-Acquisition	dB	11.4	25.3	35.4	
EIRP	dBW	5.4	29.3	29.4	From above
Losses (Other)	dB	-3.1	-3.1	-3.1	Polarization, RFI
Data/Total Power	dB	-1.0	-1.0	-1.0	Q Channel STDN 101.2
Effective EIRP	dBW	1.3	25.2	25.3	
Required EIRP	dBW	-1.7	22.2	22.3	STDN 101.2 Table 4-6
Margin-Data Detection	dB	3.0	3.0	3.0	

For SA Mode: Achievable Data Rate = 34.7 + Required EIRP
For MA Mode: Achievable Data Rate = 24.6 + Required EIRP

Actual EIRP = Required EIRP - Other Losses
Actual EIRP = Transmit Power + Passive Losses + Antenna Gain
Required Antenna Gain = Actual EIRP - Spacecraft Passive Loss - Transmit Power + Margin

Therefore: Gain = 10 Log (Data Rate) - 34.7 (or -24.6 for MA mode) = All Losses - Transmit Power + Margin

detailed definition of a hydrazine system design to provide 1040 m/sec to a spacecraft weighing 1865 kg. We would offload the tank to accommodate the specific requirements of the various TOPEX orbit options. The system provides satellite velocity, attitude positioning, and steering capability during the injection phase and coast periods associated with the selected transfer orbit scenario. The injection phase consists of a 2-burn sequence for a direct ascent trajectory from the STS orbit. Once on orbit, the Propulsion Subsystem will provide stationkeeping and backup momentum-wheel-desaturation capabilities.

TABLE 4-6. TYPICAL STDN LINKS WITH REQUIRED ANTENNA GAIN

Item	Units	High Speed Playback Mission Orbit	High Speed Playback STS Orbit	Beacon Mission Orbit	Source
Downlink Frequency	MHz	2287.5	2287.5	2287.5	Given
Transmitter Power	dBm	37	37	37	5 Watt
Filter, Switch, Cable Loss	dB	-2	-2	-2	Assumed
VSWR Loss	dB	-0.8	-0.8	-0.8	Assumed
Spacecraft Antenna Gain	dBci	-12.6	-23	-36.6	
EIRP	dBm	21.6	11.2	-2.4	
Path Loss	dBm	-175.8	-165.4	-175.8	1334 km/278 km/ 1334 km, 5° El
Polarization Loss	dB	-0.2	-0.2	-C.2	Assumed
Fading and Rain Loss	dB	-0.4	-0.4	-0.4	Assumed
Ground Antenna Gain	dB	+53.2	+53.2	+53.2	STDN 101.3 26 meter
Modulation Loss	dB	-0.7	-0.7	-0.7	Assumed
Bit Rate	dB	57	57	33	500 k, 500 k, 2 kbps
N_o	dBm/Hz	-175.9	-175.9	-175.9	STDN 101.3 LNA Preamp
Demodulator Loss	dB	-3	-3	-3	Assumed
E_b/N_o	dB	+13.6	+13.6	13.6	
Required E_b/N_o for 10^{-6} BER	dB	+10.5	+10.5	+10.5	Theory
Margin	dB	+3.1	+3.1	3.1	

Use of the STS Orbiter for launch poses additional requirements and constraints on the Propulsion Subsystem design configuration. To be certified for flight, the design must be compatible with all of the STS safety requirements as outlined in NHB 1700.7; a summary of the relevant safety requirements is presented in the Section 4.5.2, along with a statement indicating how the proposed design meet the requirement.

4.5.1.2 Delta-Launch Application

For the Delta-launched TOPEX configurations, the propulsion requirements are minimal; namely, to provide injection error orbit correction, stationkeeping, and backup momentum-wheel-desaturation capability. Based on present estimates of requirements, the most cost effective system would be a cold gas system mounted in the ESM, with velocity-correction thrusters oriented toward the deployed spacecraft CG and the $\pm$Y-axes of the spacecraft.

This TOPEX launcher option requires a significant reduction in the propulsion capability of the ATN spacecraft.

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TABLE 4-7. LINK CHARACTERISTICS SUMMARY

Link	Facility	Frequency	Modulation	Bit Rate	Value	EIRP at Spacecraft Measurement Polarization	Coverage	Spacecraft Polarization
Command	TDRS	2107.40625 MHz	QPSK	1 kbps	-148	RCP	90% of sphere	Linear
Beacon-SA Mode Operational Orbit	TDRS	2287.50000 MHz	Mod-2 added to PN code	2 kbps	5.4	RCP/LCP diversity	90% of sphere	LHC/RHC
Real-Time Science (MA)	TDRS	2287.50000 MHz	Mod-2 added to PN code	16 kbps	(24.5) 29.4	LHC	TBS cone	LHC
Low-Rate Playback (MA)	TDRS	2287.50000 MHz	As above	48 kbps	29.4	LHC	TBS cone	LHC
High-Rate Playback (SA)	TDRS	2287.50000 MHz	Q-PSK I-Mod-2 added to PN code	500 kbps	29.4	RCP/LCP diversity	TBS cone	LHC
High-Rate Playback	STDN	TBS 2300 MHz	As above	500 kbps	30.7	RCP/LCP diversity	68° cone	LHC
Beacon-Operational Orbit	STDN	TBS 2300 MHz	Mod-2 added to PN code	2 kbps	6.7	RCP/LCP diversity	90% of sphere	LHC/RHC
High Rate Playback-STS Orbit	STDN	TBS 2300 MHz	Q-PSK I-Mod-2 added to PN code	500 kbps	(20.3) 30.7	RCP/LCP diversity	68° cone	LHC
Beacon-STS Orbit	STDN	TBS 2300 MHz	Mod-2 added to PN code	2 kbps	(-1.7) 6.7	RCP/LCP diversity	90% of sphere	LHC/RHC

() = Minimum required EIRP for 3-dB margin. Larger actual value of EIRP results from gain required from same antenna when used in a different link.

4.5.1.3 Atlas-Launch Application

The Atlas-launched TOPEX configuration uses an integral solid-propellant motor to provide the final orbit-injection velocity following separation from the Atlas booster. The role of the liquid Propulsion Subsystem in this case is to provide steering and small-velocity-increment (trim) capability as well as satellite attitude positioning during the coast maneuvers associated with the

ascent mode. Once the satellite has achieved the desired orbit, the Propulsion Subsystem will provide stationkeeping and backup momentum wheel desaturation capability as in the STS configuration discussed previously.

The operational sequence through orbit injection is identical to that required of the NOAA-G propulsion system.

Although not included in this study, there may be a need to off- or on-load the solid motor to achieve the TOPEX option altitudes. It is also possible to modify the Atlas trajectory.

4.5.2 SUBSYSTEM DESCRIPTION

The proposed propulsion modules consist of combined nitrogen cold gas and hydrazine-monopropellant thruster systems for the TOPEX STS and Atlas applications and a cold gas system only for the Delta application. The hydrazine monopropellant system was selected over other options for the TOPEX orbit-transfer function based on the results of an extensive system-level trade study performed for SAATN.

In the STS and Atlas application cases, eight 2-lb_f nitrogen thrusters are mounted on the RCS support structure (RSS) periphery to provide three-axis reaction control for the coast phase attitude maneuvers and on-orbit backup momentum-wheel desaturation capability. Four hydrazine engines, with a nominal 100 lb_f thrust each, are located facing aft on the RSS. These engines provide velocity increment thrusting as well as pitch- and yaw-axis torquing for the powered flight ascent phases. The propellant and pressurant storage and feed systems are different in each case as discussed below. Two separate sets of 0.2-lb_f cold gas nitrogen thrusters and tanks mounted in the ESM are used for on-orbit stationkeeping.

A similar system comprising ten 0.2-lb_f GN_2 thrusters are used for the Delta option.

4.5.2.1 <u>STS Configuration</u>

A key element in the configuration selection was the STS safety requirement. The intent of the proposed design is to conform to all of the guidelines specified in NHB 1700.7 for liquid monopropellant propulsion systems. The resulting schematic diagram of the STS Propulsion Subsystem is shown in Figure 4-11. A summary of the applicable safety guidelines and a brief comment on how the proposed system meets these requirements are presented in Table 4-8. A list of the system components is presented in Table 4-9.

As shown in Figure 4-11, high-pressure nitrogen is stored in four titanium tanks at an initial pressure of 3600 psia. (Note that the stationkeeping system is not shown.) This storage manifold is isolated from the regulator and low-pressure nitrogen distribution manifold by a normally closed pyrotechnic valve as a safety feature for all prelaunch and STS Orbiter in-bay functions. Once the satellite is deployed in the STS reference orbit, the isolation valve is fired to the open position, allowing the nitrogen to flow through the single-stage regulator which reduces the outlet pressure to 360 psia. The

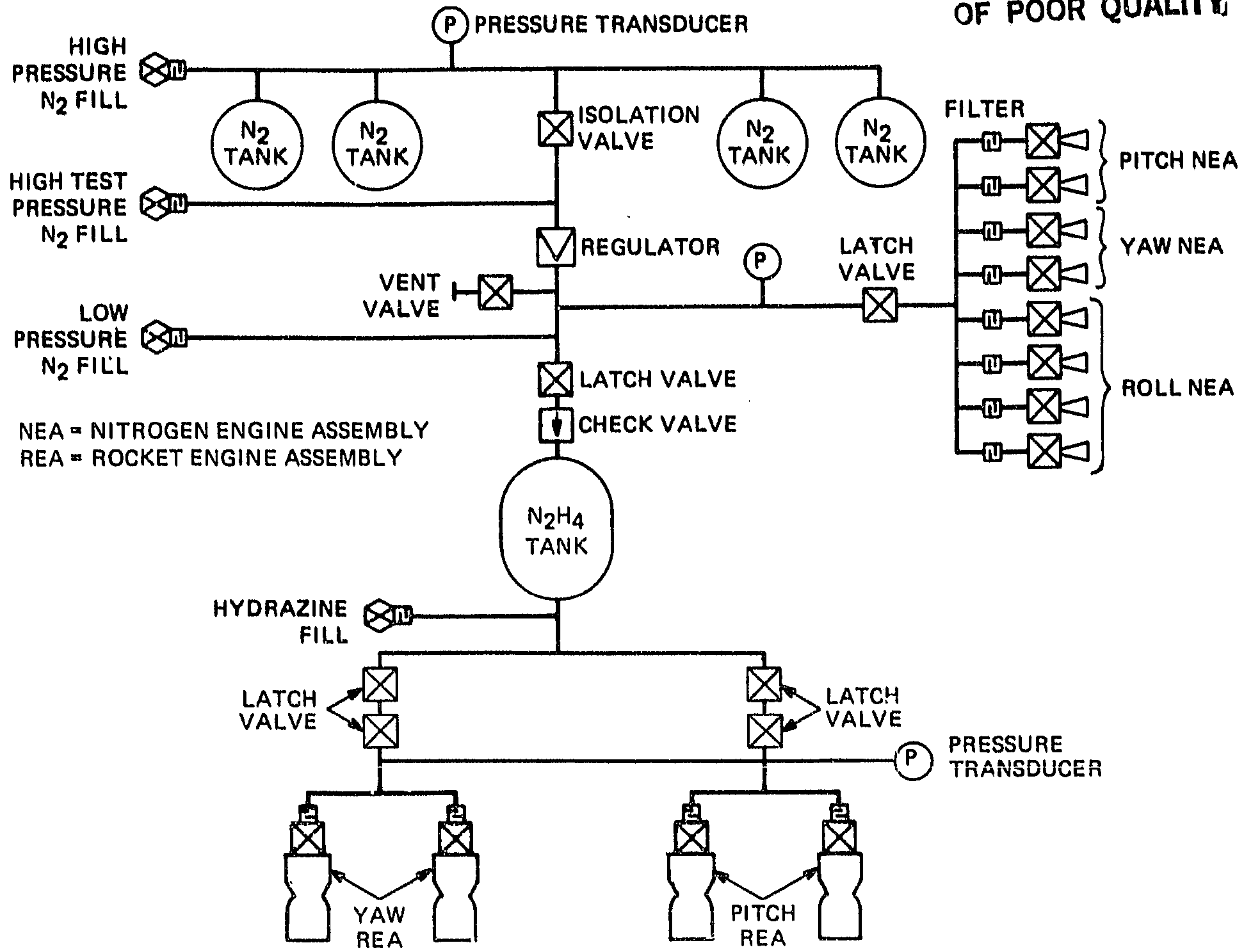

Figure 4-11. STS Propulsion Subsystem Configuration Schematic

low-pressure manifold is protected from regulator-failure overpressure by an overboard vent through a pressure-actuated relief valve. The regulated pressure supply is used for pressurizing the hydrazine propellant tank and for thrusting through the eight nitrogen thruster assemblies. The nitrogen thrusters are manifolded through a single latch valve, which is used to provide a redundant seal to the thruster valves to reduce on-orbit leakage potential. The pressurization line to the hydrazine propellant tank contains a latch valve and check valve to prevent reverse flow of the propellant into the nitrogen manifold. Three nitrogen fill valves are included in the manifold to provide entry points for pressurization and acceptance testing. Two pressure transducers are located in the nitrogen manifold; one provides monitoring of the high-pressure storage system, while the other monitors the regulator outlet pressure.

The hydrazine monopropellant is stored in a single propellant tank which incorporated fluid-motion-control baffles, but with no physical barrier between the pressurant and propellant. A surface-tension propellant reservoir is located over the outlet to provide gas-free flow to the hydrazine thrusters during zero or adverse gravity-start operations.

TABLE 4-8. STS SAFETY REQUIREMENTS, NHB 1700.7

Section No.	Requirement	How Met
202.2	Propellant delivery contested by circuiting with 3 electrical inhibits and in-flight monitoring and safing by flight crew.	(1) Valve actuation is a commanded function. (2) Electrical power to valves is from a bus that is timer activated. (3) Timer is inhibited until satellite separation. Monitoring and overrides of above are provided.
202.2B	Three mechanically independent flow control devices. Monitoring determined by safety review.	Thruster solenoid valves plus two in-series latch valves between tanks and thrusters with dry lines below tank outlet latch valves. Latch valves have position monitors.
208.4	Pressure vessels have ultimate safety factor of 1.5 and cycle life of twice expected maximum.	Qualification demonstration of design levels.
208.5	Lines and fittings have ultimate safety factor of 4.0.	Qualification demonstration of design levels.
210.1 210.2 210.3	Pyrotechnic devices meet MIL-STD-1412.	NSI or equivalent initiators will be used.
202.2	Catastrophic hazard function (namely, premature firing of pyrotechnic isolation valve followed by regulator failure) must be controlled by three inhibits.	(1) Valve actuation is a commanded function. (2) Bus power is activated by a timer. (3) Timer is inhibited prior to separation.
206	Design shall preclude failure propagation from the payload to the environment outside the payload.	The pyrovalve isolation of the high-pressure supply prevents tank failure due to a regulator failure.

TABLE 4-9. BASELINE-FEED-SYSTEM COMPONENTS SUMMARY

Component	Vendor	Heritage
Hydrazine Thruster	Rocket Research	Voyager
Propellant Tank	PSI	Viking
Propellant Tank	PSI	GPS
Pressurant Tank	PSI	Classified Program
Pressure Regulator	Marotta	TIROS-N, DMSP
Latch Valve, GN_2	Hydraulic Research	TIROS-N, DMSP
Latch Valve, N_2H_4	Consolidated Controls	Skylab
Isolation Valve, GN_2	Pyronetics	TIROS-N, DMSP
Vent Valve	Carleton Controls	TIROS-N, DMSP
Pressure Transducer	Statham	TIROS-N, DMSP, Satcom, NOVA
Check Valve	Carleton Controls	Satcom
N_2H_4 Service Valve	Pyronetics	TIROS-N, DMSP, Satcom
GN_2 Service Valve	Pyronetics	TIROS-N, DMSP, Satcom
Nitrogen Thrusters	Wright Components	TIROS-N, DMSP

The tank-outlet manifold splits into two branches, with each supplying propellant to two thrusters. One branch feeds the two yaw torque thrusters, and the other the two pitch thrusters. Each branch also contains two latch valves in series which are necessary to meet the STS safety requirements. A hydrazine fill valve is included upstream of the latch valves to permit propellant loading in the tank while the manifold remains dry below the valves. The dry manifold is required to meet STS safety requirements. Pressure transducers are located in the manifold branch lines downstream of the latch valves to verify the line-priming event and monitor the feed pressure to the engines.

All four hydrazine engines are operated simultaneously to provide the transfer orbit velocity increments. The parallel arrangement of the thrusters permits a failure-mode velocity capability with either of the two sets of engines operational. The trajectory would be altered due to the reduced thrust, but nearly the same total impulse could still be obtained from the system. Steering during velocity thrusting periods is accomplished by off-pulsing the appropriate hydrazine engine, with roll control about the thrust axis being achieved by pulsing the nitrogen thrusters. Three-axis reaction forces during coast

periods and in the final operational orbit are provided by pulsing the nitrogen thrusters. Once the TOPEX satellite reaches its desired operational orbit, all valves in the manifold system are closed. Th- hydrazine thrusters are deactivated following line residual-propellant depletion.

The pair of stationkeeping GN_2 thrusters on the ESM are oriented along the negative velocity vector in order to compensate for orbit decay.

4.5.2.2 Delta Configuration

The Delta option version of TOPEX will have eight 0.2 lb_f GN_2 thrusters on the negative velocity vector side of the spacecraft to provide orbit decay make up and backup momentum wheel desaturation. The other pair of thrusters will be oriented along the positive velocity vector and is included to provide booster injection-error correction.

The general arrangement is shown in Figure 4-12. The two tank system will operate in blowdown, with each tank capable of supporting the whole mission. Only the ΔV thrusters are provided redundantly because the other functions are backups.

The dry weight of this subsystem is 16.8 kg, and it can carry up to 16.5 kg of nitrogen.

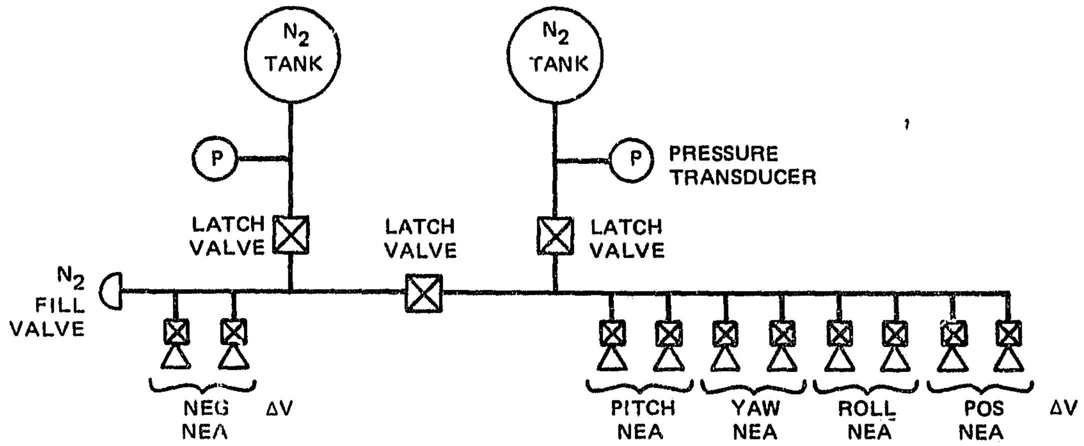

Figure 4-12. Delta Propulsion Subsystem Schematic

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4.5.2.3 Atlas Configuration

The Propulsion Subsystem schematic for the Atlas-launch application is shown in Figure 4-13 with the associated components summary presented in Table 4-9. The nitrogen is stored in two titanium tanks at an initial pressure of 3600 psia. A regulator is used to reduce the pressure of 360 psia for pressurization of the hydrazine propellant tanks and as a supply source for the nitrogen thruster assemblies. A vent valve in the low pressure manifold protects against overpressure failure in case of a leaking or failed-open regulator. A latch valve is included in the manifold leading to the eight nitrogen thrusters to provide a series redundant seal against nitrogen leakage from the thrusters during the mission life. Two manual fill valves are provided in the nitrogen manifold; one in the high-pressure side for pressurization, and one on the low-pressure side for regulator checkout and acceptance test purposes. Two pressure transducers are included to permit monitoring of both the unregulated and regulated nitrogen pressures.

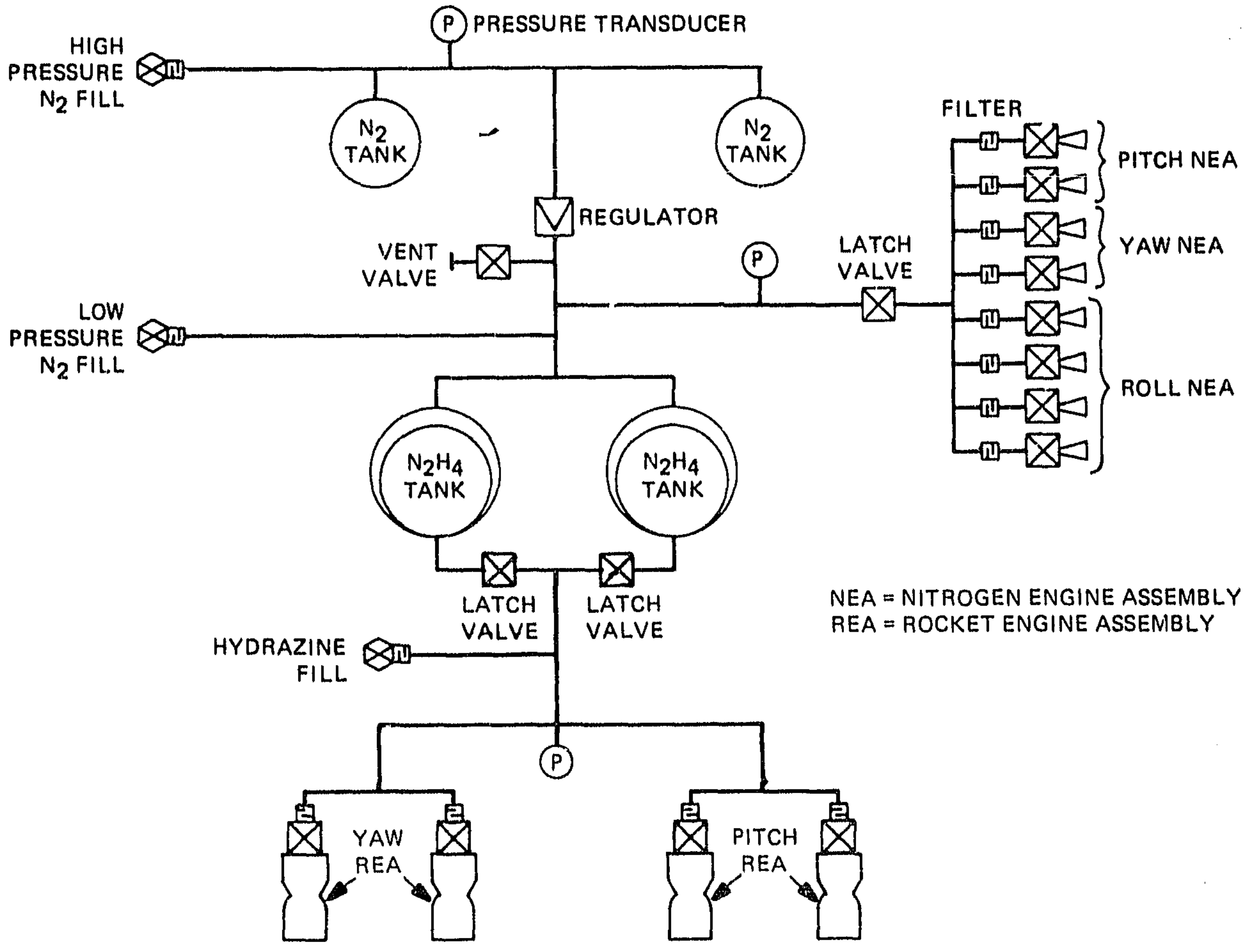

Figure 4-13. Atlas Propulsion Subsystem Configuration Schematic

The hydrazine monopropellant is stored in two propellant tanks with integral expulsion diaphragms to separate the nitrogen pressurant from the propellant. The propellant manifolds from the tanks are connected and feed two branch lines to the hydrazine thrusters; the pitch-axis pair on one side, and the yaw-axis pair on the other. A hydrazine fill valve is located in the manifold downstream of the tank outlet latch valves. This location permits equal mass filling of the two tanks by proper latch-valve operation. The two latch valves are located at the tank outlets to isolate the propellant supply from the hydrazine thruster manifold. Closing these valves permits draining of the hydrazine manifold through the thrusters after the hydrazine system functions have been completed. Line draining is performed to minimize the potential disturbance torques that might be generated by manifold rupture from freeze thaw cycling over the mission lifetime. The valves also serve as a redundant seal against hydrazine leakage from the propellant tank through the engine assemblies. A pressure transducer is included in the manifold to monitor the liquid-side pressure.

All four hydrazine engines are operated simultaneously for the Atlas booster/ATN separation and final velocity-trim maneuver. During the orbit injection solid-motor burn, the engines are pulsed individually to provide pitch and yaw steering capability. Roll control about the thrust axis is provided by the nitrogen thrusters. The nitrogen thrusters also provide three-axis attitude control during coast periods in the ascent trajectory and during the on-orbit mission life, stationkeeping, and backup momentum unloading capabilities.

The configuration of the Atlas Propulsion Subsystem is virtually identical to the NOAA-G system, except for (1) replacement of the pressurant and propellant tanks with larger capacity tankage, (2) replacement of the hydrazine pyrotechnic isolation valves at the tank outlets with latch valves, and (3) upgraded capability hydrazine thrusters (the latter to achieve commonality with the STS version).

An ESM mounted GN_2 system has been added for orbit decay makeup. It is identical with the STS option.

4.6 ELECTRICAL POWER

The TOPEX Power Subsystem will be identical to that used on the ATN spacecraft. The available payload power satisfies missions requirements with generous margins. Note that we have developed a significant number of alternate designs, for example on SAATN and MAE, to support even larger power requirements should the need arise.

Electrical power is provided by a boost discharge direct energy transfer (DET) system (see Figure 4-14). The primary power source is a single-axis-oriented solar array, and the secondary source is a set of three nickel-cadmium batteries. The Power Subsystem output is regulated $\pm$28 Vdc and regulated +5 Vdc. (Plus and minus 10 volts are provided by the CIU.) The major components are the solar array, batteries, power supply electronics (PSE), battery charge assembly (BCA), power converter (PC), solar array drive (SAD), and array drive electronics (ADE).

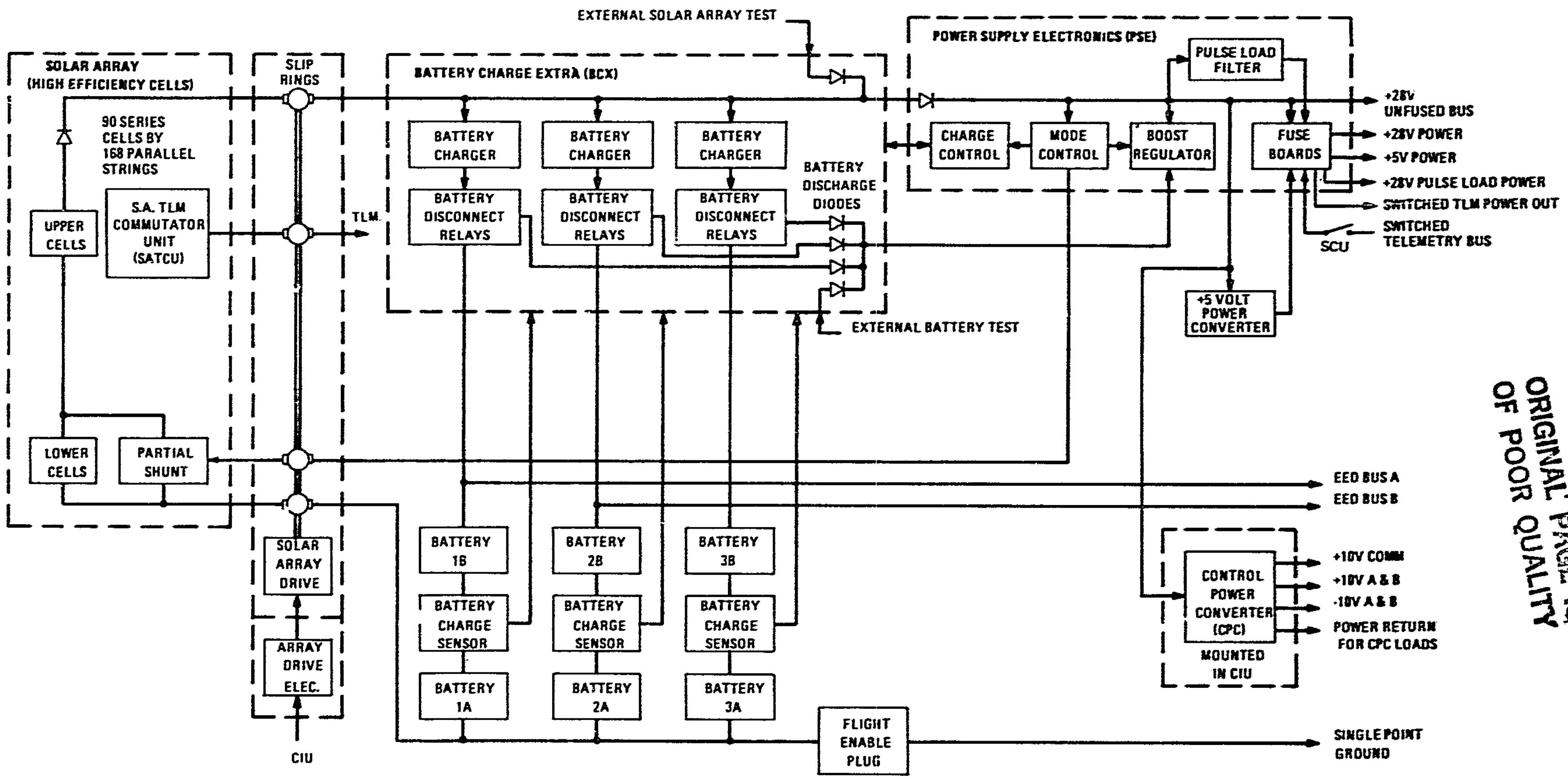

Figure 4-14. SAATN Power Subsystem, Block Diagram

In the mission mode, the SAD (controlled by the ADE) rotates the solar array (which on ATN is canted at 36 degrees to the orbit normal) once per orbit so that it continually faces the sum. The cant angle may be optimized for a given orbit's altitude and sun angle range. For example, on DMSP the sun angle range is 0 to 90 degrees compared to 0 to 68 degrees on ATN. The optimum cant angle on DMSP is 30 degrees. The optimum for TOPEX will be closer to that of DMSP than that of TIROS and is easily accommodated by adjusting the cant offset angle. In order to maintain a resonable sun-angle range, the spacecraft must be maneuvered so that the array drive axis is parallel to the orbit normal, on the same side of the orbit as the sun. This "flip maneuver" is discussed in Section 4.2 and is a conventional requirement for DMSP.

The solar array supplies current through slip rings in the SAD to the PSE during normal daytime operation. Power above that required by satellite loads and battery charging is dissipated by partial shunts which are located on the array so as to dissipate the excess power outside the main modules of the satellite. Total orbit-average load capacity for the system is forecast as a minimum of 539 watts (end-of-life) after 2 years. The three batteries (each having a capacity of 26.5 ampere-hours) supply power through the boost regulator during the dark portions of each orbit and augment the solar array for peak-load conditions during daylight portions.

Each battery consists of two battery packs. A mode controller senses the +28-Vdc regulated bus voltage and operates the partial shunts and/or charge regulator as required. The power converter derives +5 Vdc regulated power (which is used to power interface circuits) from the +28-Vdc regulated power.

Automatic switchover occurs from primary to backup circuitry for the boost regulator, charge regulator, and mode controller in response to signals from failure-detection circuits. Either primary or backup circuits may be selected by ground command. Commandable battery charge-and-discharge disconnect relays are provided. Full circuit redundancy also exists in the power converter, ADE, and partial shunts.

The TOPEX load requirements by subsystem and specified option are summarized in Table 4-10.

The capacity of the ATN power system to accommodate electrical loads is specified by GSFC 480-5B to be a minimum of 506.5 watts after 2 years in a 450 nmi circular orbit. The forecast minimum varies from 515 to 539 watts depending on which V-T limit curve is used. Thus, we have a margin from 123 to 235 watts depending on options and assumptions. Figure 4-15 shows even further power available as a function of sun angle. As noted before, the specified operating range of ATN is 0 to 68°, whereas the range is 3.5° to 90° for TOPEX. With the 36° cant angle of the ATN array, we would merely continue the curve to the right. However, by decreasing the cant angle, we can raise the right-hand side while losing somewhat on the left side (the side approaching 0° sun angles). The detailed tradeoff remains to be done.

The hardware elements of the Power Subsystem are summarized in Table 4-11.

TABLE 4-10. TOPEX ORBIT AVERAGE LOAD REQUIREMENTS

Item	Load (watts)
Bus	
Attitude Determination & Control	53.5
Command and Data Handling	45.9
Telecommunications	12.5
Thermal (varies with sun angle)	77.0
Power (SAD and ADE)	5.9
Bus Subtotal	194.8
Payload	
Option 1	259.0
Total	453.8
Option 2	180.0
Total	374.8
Option 3	200.0
Total	394.8

4.7 AIRBORNE SUPPORT EQUIPMENT (ASE)

The major item of ASE is the SSV cradle. The electronics to interface with SSV checkout systems, checkout software, and harness for the cradle are all required and sized based on our SAATN work (and are included in the cost estimates). On the other hand, the cradle, which can be a major cost driver to the program, has certain options which should be considered. There is at present a high probability that TIROS or DMSP will be launched by the SSV before TOPEX. If this happens, all SSV interface problems will be solved, and all cradle and other interface hardware will be available for use by TOPEX, subject to agreements with NOAA and/or the Air Force to use the TIROS (or DMSP) ASE. At present, however, it is not certain that any of this hardware will be available in the TOPEX time frame. Therefore, as a worst case scenario, we have assumed that the TOPEX project will be responsible for providing all of the ASE for the mission.

In this case, there are two further options; (1) use the existing (or slightly modified) ASE from some other project (or upper stage), or, (2) build a new version of the ASE. At the onset of the SAATN study, a tradeoff was made between utilizing a modified version of the PAM-D cradle/cocoon, the basic cradle, in a "stretch" version and designing a TIROS-unique cradle. MDAC was to be used as a major subcontractor in both cases and, based on in-house and MDAC analysis, it was determined that the costs for the two were equivalent, therefore, the decision was made to opt for the new version.

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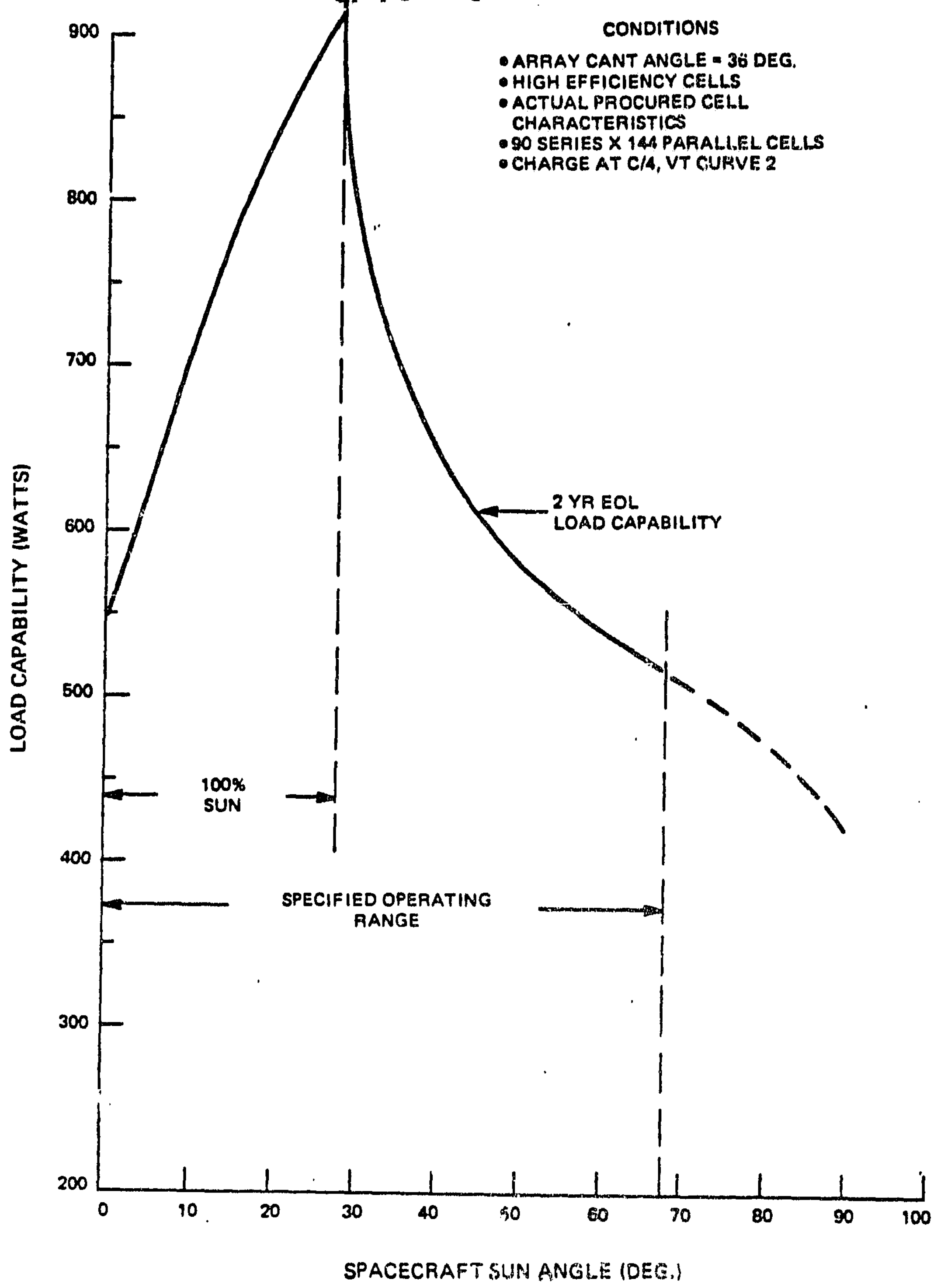

Figure 4-15. EOL Load Capability Versus Sun Angle

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Table 4-11. EPS COMPONENT CHARACTERISTICS (ATN)

Parameter	Value
SOLAR ARRAY (SA) (RCA 2297950)	
Panels per Spacecraft	8
Panel Area Dimensions	93.5 x 24.2 inches
Panel Weight	14.3 pounds
Total Array Area	125.6 square feet
Total Array Weight	114.4 pounds
Solar Cell Characteristics	
Type	High Efficiency
Area	2 x 4 cm (oversized)
Base Resistivity	1-3 ohm-cm
Thickness	0.010 inch
Coverglass	
Type	Fixed Silica
Thickness	0.006 inch
Array Solar Cell Layout	
Number of Series Cells	90
Number of Parallel Strings	144
Partial Shunt Configuration	
Number of Partial Shunt Circuits	48 (6 per panel)
Number of Series Solar Cells Across Each Shunt	58
SOLAR ARRAY DRIVE UNIT (SAD) (RCA 2276893)	
Motor Type	DC Brushless Torque Motor
Number of Power Sliprings Used	16
Number of Signal Sliprings	7
Weight	10.2 pounds
SOLAR ARRAY DRIVE ELECTRONICS (ADE) RCA 2284574)	
Weight	5.8 pounds
Size	6.7 x 5.1 x 8.7 inches
BATTERY (BAT) (RCA 2276849 & RCA 2276850)	
Number per Spacecraft	3
Number of Series Cells	17
Number of Assemblies	
9-Cell Pack	3
8-Cell Pack	3

Table 4-11. EPS COMPONENT CHARACTERISTICS (ATN) (Continued)

Parameter	Value
BATTERY (BAT) (RCA 2276849 & RCA 2276850) (Continued)	
Weight of Assemblies	
9-Cell Pack	28.8 pounds
8-Cell Pack	25.9 pounds
Total Battery Weight	164.1 pounds
Pack Size (Both Packs Identical)	[illegible] x 9.1 x 5.5 centimeters
Charge Control System	
Current Limiting Selectable by Command	10, 7.5, or 0.5 Amperes
Voltage Limiting (Temperature Dependent)	4 Levels by Command
POWER SUPPLY ELECTRONICS (PSE) (RCA 2297292)	
Weight	26.2 pounds
Size	16.1 x 10.8 x 7.7 inches
+28V Bus Characteristics	
1. Regulation	± 2 percent
2. Ripple	Less than 50 millivolts (into a resistive load)
3. Transient Response	0.2 volt (0-p) for load change of 6 amperes at a rate of 20 mA/us
Boost Regulator Efficiency	87 percent minimum (at full load)
BATTERY CHARGE ASSEMBLY (BCX) (2297289)	
Weight Radiator Plate	} 17.1 pounds
Size	25.0 x 18.0 x 3.0
Charge Current Range per Battery	0.5 to 10.0 amperes
POWER CONVERTER (PC) (RCA 1972743)	
Weight	1.1 pounds
Size	4.0 x 4.0 x 2.5 inches
+5V Bus Characteristics	
1. Regulation	5 percent
2. Ripple Voltage	Less than 50 millivolts (into a resistive load)
Converter Efficiency	77 percent minimum (at full load)
BATTERY CURRENT SENSORS (BCS) RCA 2294097)	
Number per Spacecraft	3
Weight (each)	1.0 pound
Size (each)	4 x 7.3 x 1.7 inches

During the TOPEX study, this decision was reevalutated briefly for two reasons. First, we thought there might be some advantage if we could find a cradle that could be used with less modification than the PAM-D design would have required. Second, although the TIROS spacecraft requires a cocoon for the SSV launch for contamination controls, it is not clear that the TOPEX instrument complement requires the same degree of cleanliness. This is important because the PAM-D version of the cradle always comes with the cocoon, and the cocoon is an expensive item.

We briefly contacted the TDRS PMO at GSFC to inquire how they are solving the SSV accommodation problem. They are using the two-stage IUS, and all of their SSV interface problems are being handled by Boeing as part of the IUS. In essence, this resolves their problem by elimination of this area as a PMO responsibility, at least from an accounting point of view if not from a technical one. This is a significant point for TOPEX in that it may affect the overall cost chargeable to the project. We considered the possibility of literally using the IUS cradle without the IUS with a unique adapter between the IUS adapter fitting and the spacecraft. This looks viable; however, we did not address the problem of landing loads in the cantilevered position, and a more detailed study would be required to do so. We do know that this is the configuration that TDRS will fly in on the IUS, cantilevered on an extremely large stack (at least 15 to 20 feet longer than the TOPEX stack); therefore, it is not unreasonable to assume this problem has been addressed. IUS and/or TDRS project discussions are most likely where a more detailed study would begin. Finally, TDRS has opted not to use a cocoon in the SSV, and access to the environmental analysis that led to this decision would be helpful in determining TOPEX requirements.

The primary conclusion that can be drawn at this time is that the best approach to the SSV ASE for the TOPEX mission is still not clear, and further study needs to be done before the tradeoffs can be clearly evaluated and priorities established. Further, the primary driver of the tradeoff will be cost since a variety of alternatives are technically feasible.

4.8 TOPEX AEROSPACE GROUND EQUIPMENT (TAGE)

The Ground Support Equipment (GSE) comprises all the mechanical and electrical equipment required for handling and testing the TOPEX satellite from bus assembly through launch. The GSE will consist of a modified Advanced TIROS-N AGE (ATNAGE), launch-site support equipment, and assorted handling and support fixtures. A photograph of the existing ATNAGE is presented in Figure 4-16. The ATNAGE hardware and software configurations including the changes required for TOPEX are described herein. A description of the fixtures and launch-site equipment requirements is also presented.

4.8.1 SYSTEM CONFIGURATION

The baseline TAGE system will consist of three Data General computers and associated peripherals. Although the existing ATNAGE may be available for TOPEX, depending on the state of the TIROS project requirements at that time, we expect that this is unlikely and would recommend an updated computer to replace the old Data General (DG) S/200 systems. We would, however, retain the same basic system structure and data flow. Note that the TIROS data stream is

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Figure 4-16. TIROS-N Aerospace Ground Equipment

significantly more complex than that of TOPEX, due to the number of telemetry streams, the type of data, and the rates. Significant simplification may be possible. To illustrate this, we show the ATNAGE in Figure 4-17. The three computers are designated as Computers A, B, and C and are used as follows:

(a) Computer A is a DG S/200 dedicated to running high rate information (HRI) software analysis. All control comes from, and all analysis results go to, Computer B.

(b) Computer B is also a DG S/200. The satellite bus and system control software resides in the foreground of Computer B. All telemetry ingest, TIP telemetry limit checking, command verification, and command generation take place in this computer. Computer B also sends low-rate data to Computer C and receives messages from it. Computer B also controls and receives messages from Computer A. All operator control inputs go through Computer B. The Atlas run-time system is used in the performance of all Atlas-controlled test procedures and resides in the background of Comp ter B.

(c) Computer C is a DG S/230. All low-rate instrument (LRI) software analyses for the MSU, SSU, DCS, SEM, HIRS, SAR, ERBE-S, ERBE-NS, and SBUV instruments are performed in Computer C.

Above each computer in the figure is shown its primary function in the TOPEX system. Computer A will act as a buffer for data acquisition at high rates and for data archiving; this permits the collection of data to proceed independent of the load on the rest of the system. Computer B will be able to interface directly with the spacecraft to acquire low-rate data (1 to 2 kbps definitely and the

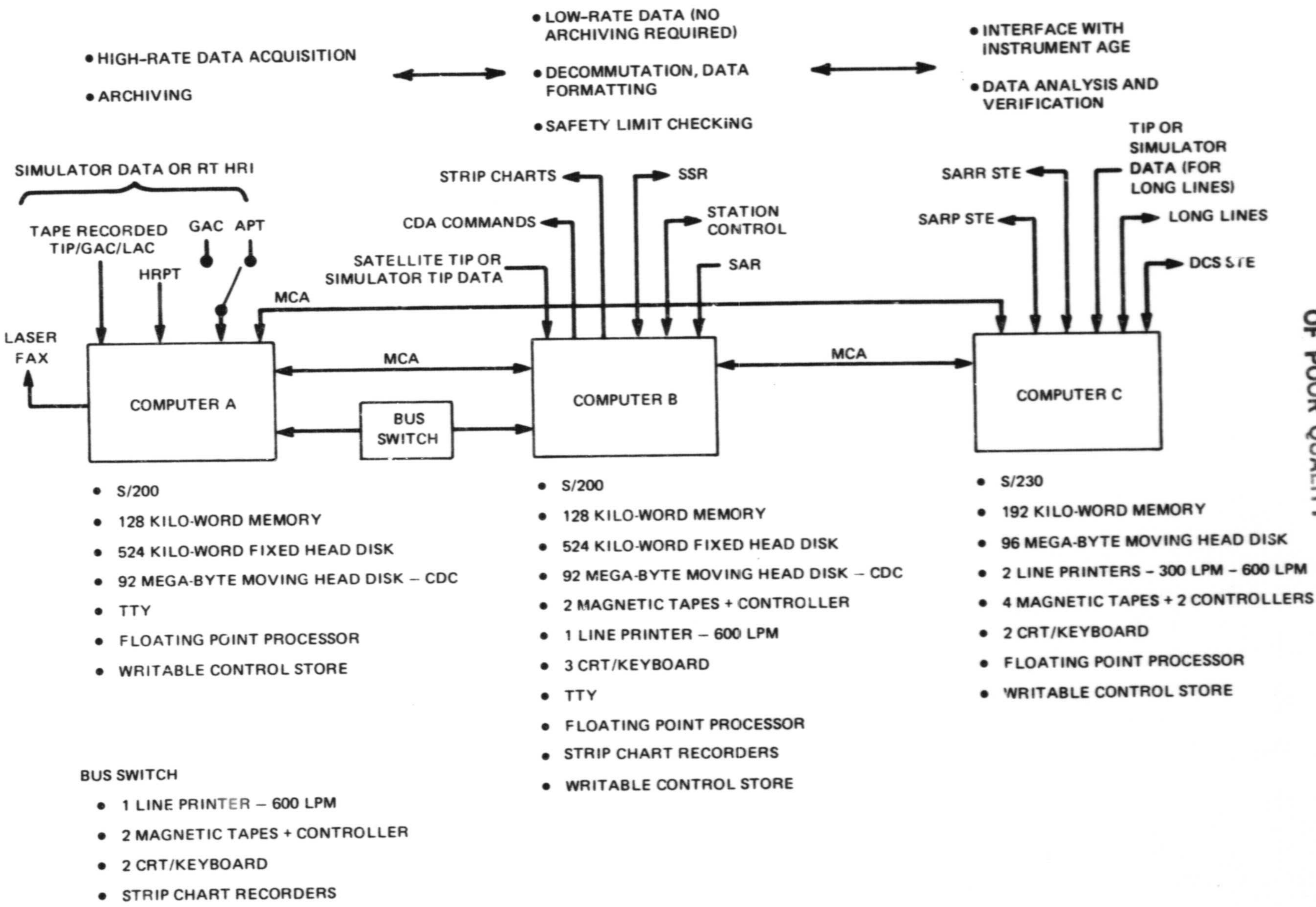


Figure 4-17. ATNAGE System Configuration with TOPEX Modifications

∿16 kbps data depending on computer load) to process in real-time. This computer will be real-time only and, if it cannot keep up with the data stream while performing certain functions, it will buffer itself via computer A and acquire the data from A as it becomes ready to process it. Computer C will be the prime machine for data analysis and evaluation for the spacecraft and will interface with the instrument AGE systems. Note, finally, that if the DG machines are replaced, we will evaluate the possibility of combining two or even all three of the computers into a somewhat larger machine. However, even in this case, the three conceptual functions described above will be maintained, and the interface will be via internal software rather than hardware links between machines.

4.8.2 HARDWARE CONFIGURATION

The ATNAGE configuration is shown in Figure 4-18 as modified for TOPEX. The ATNAGE computer hardware currently consists of a series of Data General Eclipse S/200 and Eclipse S/230 computer systems, with general-purpose input-output (I/O) boards to handle TIP and HRI processing, and other special-purpose processing. These general-purpose boards were designed and developed at RCA for use in the Data General computers. There are two identical computer systems used in the testing of satellites at RCA Astro-Electronics. One of the computers is shipped to WTR and used in the monitoring and commanding of the satellites prior to the launch. There are two additional computers in the ATNAGE system; one is used as a satellite simulator, and the second is the Software Development Facility (SDF). The simulator is an Eclipse S/200 which is primarily used in the testing of the software and the resolution of these discrepancies. The SDF is used for the compilation and basic module testing of the software, data base maintenance, and maintenance of the Atlas test procedures.

The ATNAGE hardware complement also includes the following:

(a) The Satellite Support Rack (SSR) and Remote Power Switch (RPS) shown in Figure 4-19 include all the hardware required to excite and monitor satellite hard-line test inputs, including power and to monitor non-mission-data hard-line test outputs.

(b) RF processing equipment includes all the receivers, RF switches, bit synchronizers, frame synchronizers, command transmitters, command generators (both computer controlled and manual controlled), and recording and monitoring equipment necessary to test the satellite. The RF section of the ATNAGE is shown in Figure 4-20.

(c) The Ordnance Device Simulator (ODS) is a portable, manually operated unit (see Figure 4-21) built to meet safety requirements for operation with the satellite on the booster and is capable of interfacing directly with the satellite during testing.

4.8.3 SPECIAL TOOLS AND FIXUTRES

Much of the fixturing developed during the TIROS and Advanced TIROS programs will remain usable for the TOPEX program. The new fixturing that is required

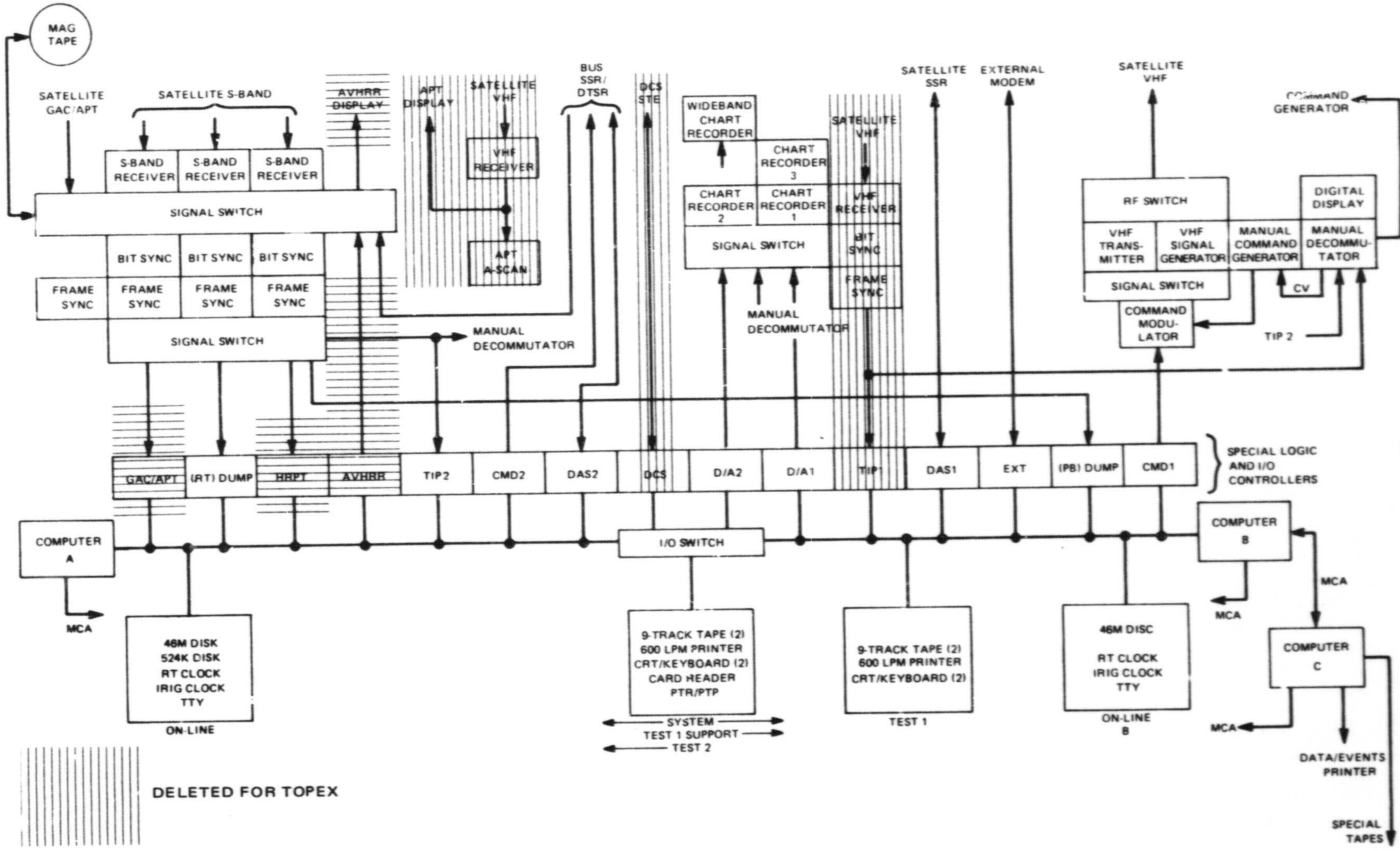
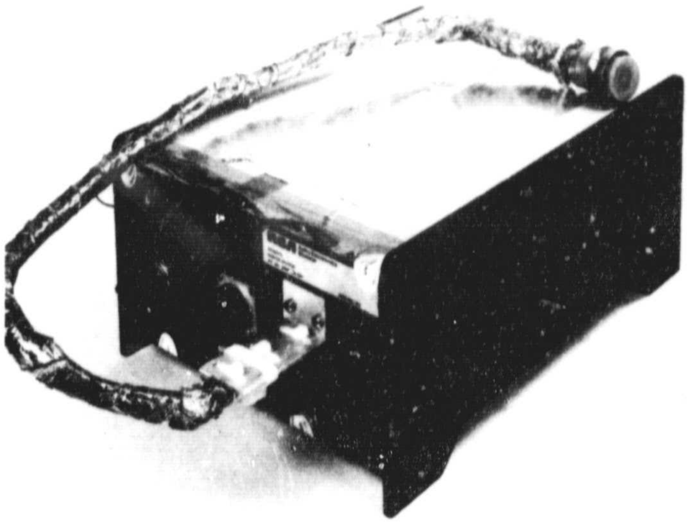


Figure 4-18. ATNAGE Hardware Configuration



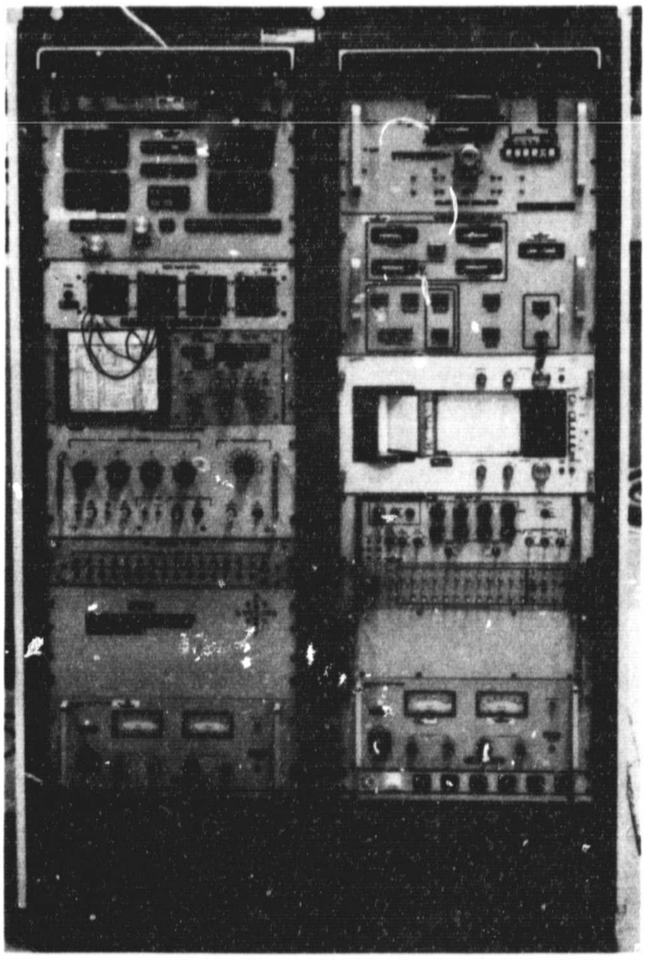


Figure 4-19. Satellite Support Rack and Remote Power Switch

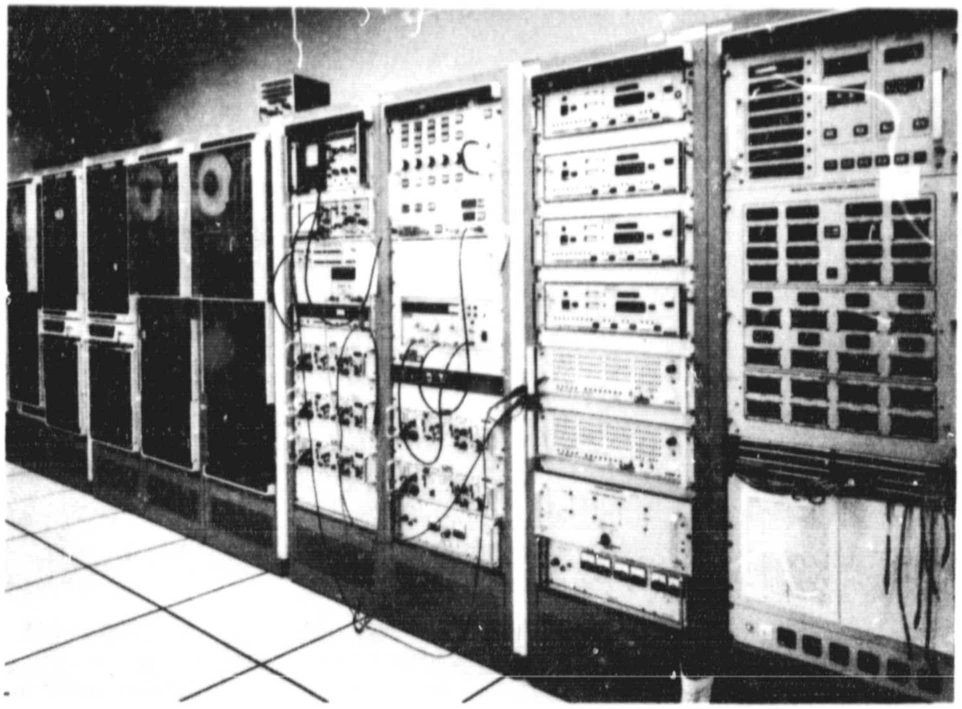


Figure 4-20. RF Section of ATNAGE

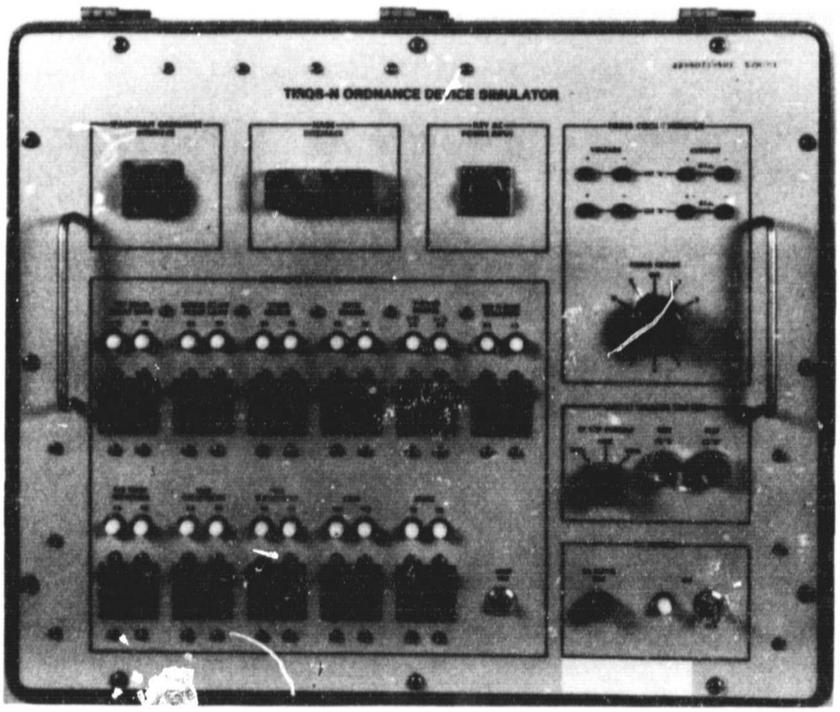


Figure 4-21. Ordnance Device Simulator

is necessitated in general by:

- Box additions and/or changes in the ESM and the deletion of the IMP.
- Change to the propulsion system.
- A Shuttle-mounting arrangement totally different from the ELV.

The new fixtures are required for box changes and/or addition, and for a generally heavier satellite. In particular, the following new fixtures will be required:

- ESM panel rearrangement will require new honeycomb panel routing templates and drill jigs.

- Manufacturing tooling will be needed to wind and form the new front-panel torquing coil.

- Modification to the satellite thermal-vacuum fixture will be required to provide the new earth sensors (for STS launch options) with cold targets.

- Because of the weight increase of the overall satellite, a change in the cross section and profile of the V-band separation rig is anticipated. New RSS interface adapters (used during RCE buildup, boom deployment, and mass-properties measurements) will be required.

Fixtures required because of the new propulsion system (for STS launch options) are as follows:

- The ATN RCE mockup will be upgraded to represent STS/Atlas. This fixture is used during development of manifolds and bracketry for the propulsion system.

- A tank handling and transport fixture will be built for storing, transporting, and installing the new hydrazine tank.

New fixtures (for STS launch options) required because of the mounting arrangement in the Shuttle, a three point pick-up along-X (satellite coordinates), are as follows:

- Interface fixtures, two types, representing the intermediate-cradle-to-cradle interface and the satellite-to-intermediate-cradle interface.

- Dummy separation bolts.

- A horizontal lifting fixture for lifting the satellite in the +X up attitude.

- A spring caging tool, used for compressing the separation springs before mating and demating the satellite from the intermediate cradle.

- A modified shipping container capable of interfacing with the intermediate cradle.

SECTION 5.0

COST ANALYSIS

SECTION 5.0
COST ANALYSIS

5.1 COST ANALYSIS BACKGROUND

A budgetary estimate has been developed for the TOPEX satellite based upon the preliminary baseline design described in Sections 3.0 and 4.0. Data was used from the past 6 years of TIROS ATN program cost history as a basis for the task estimates. The TIROS program uses a conventional expendable launch vehicle (Atlas E/F). A series of studies were conducted during 1979 and 1980 to determine the changes necessary to adapt the TIROS ATN spacecraft for launch by the Space Transportation System (STS). The necessary design modifications, system analysis, and interface requirements were identified, and a formal quotation was submitted to NASA for the total program in response to an RFP for spacecraft hardware. Due to the thorough system understanding and task identification derived as a result of the study and proposal efforts, the costs identified for the Shuttle-launched TIROS spacecraft were used where applicable to derive the budgetary estimate for TOPEX.

5.2 PROGRAM SCHEDULES

The TOPEX programmatics are shown in Figure 5-1. This is a plan for a launch date 46 months after program start on an STS-launched vehicle. This was selected as the baseline for planning and costing because it would be the most constraining relative to the existing TIROS program. If TIROS or DMSP contract for an STS launch prior to TOPEX (which is likely), the schedule could be reduced to 36 months, and there would be significant cost reductions. Typical phasing for the program activities is shown in Figure 5-1 for the major subsystem deliveries. A thumbnail sketch of the program reveals the following highlights.

5.2.1 PHASE 1 - SYSTEM SPECIFICATION REVISION

Prior to start of major design activities, the governing specifications for subsystems and components, design, and test requirements are reviewed and revised as necessary to reflect the TOPEX revisions from the TIROS configurations. Particular emphasis is placed upon items procured from vendors, designs which use long-lead materials, and functional areas which have multiple and complex interfaces. During this phase, the operating Program Management Office will be staffed, and the program operating instructions and administrative controls will be derived. To reduce the program lead time, this activity can be accomplished during a pre-award phase which could be a Phase A study effort.

5.2.2 PHASE 2 - DESIGN

The program design phase will take place during months 4 to 11. A Preliminary Design Review is scheduled in month 4. During the PDR, the following accomplishments will be reviewed:

- Selection of design approaches from tradeoffs in areas of new design.

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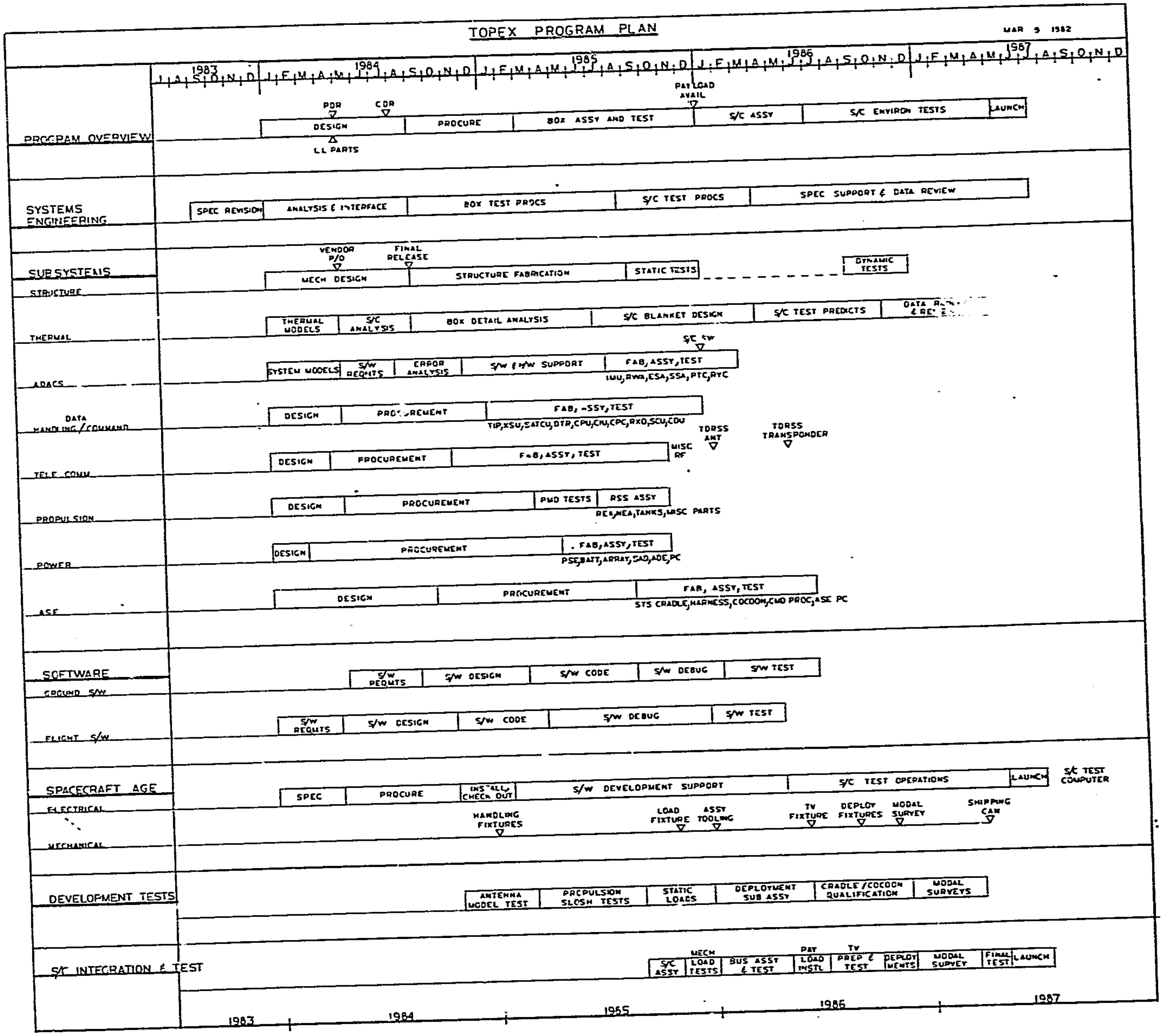

Figure 5-1. TOPEX Program Plan

- Confirmation of applicability of existing and modified TIROS ATN designs.

- Identification and approval of release for long-lead material.

- Confirmation of system performance through analyses and error budgets.

A program Critical Design Review (CDR) will be held in the 12th month of the program, 9 months after the start of the design phase. At the completion of the CDR, all designs will be completed, and remaining parts will be procured.

5.2.3 PHASE 3 - PROCUREMENT, FABRICATION, ASSEMBLY AND TEST

The procurement phase, which starts with the release of long-lead parts and vendor subcontracts, will be the predominant activity between months 12 and 24. In addition to the engineering support required for the manufacturing operations, special test equipment hardware and software will be developed to accomplish the test phase during this period. Development tests required to qualify hardware or to prove design performance on a one-time basis will be accomplished prior to start of spacecraft assembly. All components and boxes will be subjected to qualification or acceptance testing prior to next higher assembly integration.

5.2.4 PHASE 4 - SPACECRAFT INTEGRATION AND TEST

This phase of the program will bring together the hardware, software, test equipment, and procedures developed for the TOPEX spacecraft. The mechanical portions of the spacecraft will be assembled and static load tested. All bus boxes will be installed and tested, demonstrating performance compliance and compatibility prior to integration of the payload. After installation of the payload and completion of satellite level performance tests, the spacecraft will be subjected to environmental exposures, designed to demonstrate system compliance with the launch and orbital exposures anticipated for the satellite.

5.2.5 PHASE 5 - LAUNCH, ACTIVATION, AND EVALUATION

At the successful completion of the Integration and Test Phase, the spacecraft and ancilliary equipment will be shipped to the launch site, and full satellite electrical performance tests will be completed prior to launch. After orbit insertion, deployments, and completion of activation and evaluation, the program will terminate.

The programmatic flow and timeline described is similar to that used repetitively on the TIROS program. Full confidence exists on the ability to complete all series and parallel tasks necessary for successful program completion within the budgets established for the efforts.

5.3 COST ASSUMPTIONS

In order to derive a valid and accurate cost estimate and, at the same time, be independent of variables not under direct control of the TOPEX program, reasonable assumptions were made concerning the relationships of the TOPEX to

the TIROS programs. A degree of interdependence and independence was assumed such that maximum advantage could be obtained from the TIROS program experience, and the responsiveness to the unique requirements of the TOPEX program will be assured via independent program control. The TOPEX program was modeled using the TIROS/DMSP program relationship developed successfully since the inception of the two programs. The TIROS and DMSP programs are responsive to two separate customers with similar designs. The programs share designs, equipment, and personnel where appropriate, yet maintain separate program control and resources where necessary. The following program assumptions were made:

- A separate Program Management Office, system team, and integration and test activity were assumed for TOPEX. All personnel designated as program cadre will be dedicated to the TOPEX program.

- Full use will be made of designs, analysis, reports, procedures, etc., developed for TIROS or DMSP where applicable to TOPEX. All such design and test documentation will be modified and reidentified and maintained in a TOPEX configuration management system.

- All test equipment, tooling, fixtures, etc., required on a dedicated basis will be acquired as part of the TOPEX program costs. Fixtures, tooling, and other special test equipment or resources used on an infrequent basis will be shared on a noninterfering basis as presently practiced by the DMSP/TIROS programs.

- The program organization is based upon a matrix concept, where engineering skill group expertise is used for design and development of unique subsystems. Other RCA Astro-Electronics departments will supply manufacturing, financial, reliability, and procurement support as required during the various phases of the program. A TOPEX Program Management Office will be established to provide overall management, direction, financial control, customer interface, and liaison across all facets of the program. A dedicated Program Manager will be totally responsible for all engineering, financial, and management aspects of the program.

- All nonrecurring development costs relating to Shuttle launch accommodation is assumed to be borne by the TOPEX program. Although the TIROS program was instrumental in the study and identification of Shuttle-related tasks, the current TIROS program is scheduled to be launched on expendable boosters; therefore, the tasks described have not been accomplished. Should the DMSP program or any other applicable experience be available at the time of TOPEX program implementation, the costs can be reduced or shared, reducing the costs given in this budgetary estimate. To provide an assessment of the potential (and likely savings for Shuttle design costs), they are identified separately in the cost matrix.

- All nonrecurring costs necessary to replace obsolete or otherwise nonobtainable parts used on the original TIROS design developed during the late 1970's are assumed to be borne by the TIROS program. The timing of TOPEX will allow full advantage to be taken of the technology update planned by the TIROS Program for TIROS-H, -I, and -J.

- A full set of critical spare parts and board assemblies will be supplied. This spares philosophy is the same as proposed for the Shuttle-launched ATN program. The availability of critical parts and assemblies maintains the ability to provide a quick turnaround in case of malfunction, and avoids the costly replication of all boxes on the program. The spare board assemblies will be tested and qualified at the board level to allow substitution late in the test cycle without costly or time consuming requalification.

- All costs are given in constant 1982 dollars, no inflation adjustments are included.

- Quantities of items required are given in Table 1-2. No nonflying prototype or special test model spacecraft level or component level test models are required with the exception of the components shown. The flight boxes which are new or modified extensively will be tested to protoflight levels and flown on the spacecraft. Structure static load tests and modal surveys will be conducted on the flight structure nondestructively. Thermal performance will be demonstrated on the flight spacecraft during the thermal-vacuum test cycle.

- Parts used in construction of the spacecraft will be assumed to be acceptable for use on TOPEX if previously approved for use on TIROS. All new parts will be selected in accordance with the RCA-approved parts list.

APPENDIX A

RCA RESPONSES TO JPL MEMO TPO 82-81

1. SYSTEM ENGINEERING

QUESTION 1A: What is the current development status of ATN (as opposed to that of the components)?

RESPONSE 1A: The ATN spacecraft is completely integrated with all flight components and instruments installed. The spacecraft has successfully completed both an electrical performance and evaluation testing and a 21-day thermal-balance and thermal-vacuum test.

Following the successful completion of a baseline electrical performance and evaluation test, the environmental qualification program began. The first environmental test was an EMI compatibility survey performed in an anechoic chamber. The individual characteristics of each spacecraft transmitter and receiver as well as the effect of various combinations of these on other spacecraft systems were evaluated. All spacecraft antennas were installed in their orbital configuration for this test. The results were consistent with the expected design requirements. Following the EMI test, a combined 21-day thermal-balance and thermal-vacuum test was performed. This test both validated the thermal design of the spacecraft and demonstrated the ability of the components to survive thermal cycling. Being a qualification test, the temperature plateau extremes were 10°C beyond the flight predictions.

Following the thermal-vacuum test, the integrity of all spacecraft systems was demonstrated during an electrical performance and evaluation test. The vibration testing phase is scheduled to begin during the week of May 10, 1982. Prior to sinusoidal vibration testing, all deployable appendages will be demonstrated by manual release. At the completion of sinusoidal vibration, the spacecraft will undergo acoustic testing. All vibration levels will be at protoflight levels (sine - 1.25 times flight; acoustics 3 dB higher than flight). The deployable appendages will be pyrotechnically released and tested at the completion of acoustic testing.

After another electrical performance and evaluation test is performed to verify all spacecraft systems, the spacecraft will be readied for storage in preparation for shipment to the launch site.

QUESTION 1B: What is the assesment of suitability of ATN for the 5-year TOPEX extended mission?

RESPONSE 1B: Recently, RCA Astro-Electronics signed a contract with the Air Force Space Division for an extension of the DMSP mission to 4 years. This contract included negative incentives as a function of spacecraft life. Our most significant concern was the increase in total dose radiation. We found that the CPU memory was the most sensitive and have specified a more hardened component to the memory supplier. On digital devices, RCA has recently demonstrated hardness to transient radiation and total dose (equal to 10^6 rads). The solar arrays were resized to satisfy the requirements of the DMSP mission. We have run tests on moving mechanical assemblies (i.e., the ACS wheels and solar array drive) and have demonstrated 5-year lifetimes. Finally, the extension to a 5-year mission was found to have an insignificant effect on costs.

On TOPEX, the costs submitted with the draft Final Report cover an extension to a 5-year mission. The combination of the extended life and higher altitude may have a minor impact on costs due to the need for additional shielding and/or special radiation hardened parts.

QUESTION 1C: Are the payload sensors assumed to conform to a standard set of interfaces? If so, what are those interfaces?

RESPONSE 1C: We have assumed that the TOPEX program could accept the standard TIROS interfaces. The TIROS General Instrument Interface Specification (submitted to JPL on May 7, 1982) is used as a baseline for negotiating what we call a Unique Instrument Interface Specification which would be a separate document for each of the TOPEX payloads and which would be the governing interface specification.

2. ATTITUDE DETERMINATION AND CONTROL

QUESTION 2A: Describe the satellite nadir pointing concept; hardware implementation; and equipment sensitivity to gravity gradient, drag, and solar pressure.

RESPONSE 2A: For Delta/Atlas launches where the passive Earth Sensor Assembly (ESA) is used, nadir pointing is realized by achieving a balanced condition among the radiation inputs received by the thermopile detector elements within each of the four assembly viewing windows. The lines of sight of the viewing windows lie on the surface of a cone, whose central axis coincides with the nadir directed body axis, symmetrically located at 90° intervals of azimuth displacement. The indicated geometry provides quadrature segment sensing of the earth's limb.

For STS launches where the active Conical Scanning Sensor (CSS) is used, nadir pointing is realized by achieving a balanced condition between the earth chord lengths developed by the cross track looking scanner (i.e., up and down, directed along the orbit normal vector) and by splitting (i.e., bisecting) the individual chord lengths relative to a fixed reference coincident with the projection of nadir directed body vector in the scan cone base plane.

More detailed descriptions of the ESA and CSS are provided in Exhibits 1 and 2, respectively.

The integrated effects of environmental disturbances are accommodated by the nominally "zero momentum" on-orbit control system via momentum storage in the Reaction Wheel Assemblies (RWA's). Wheel momentum unloading at modest thresholds, typically less than 10 in/lb-sec/axis, is accomplished magnetically with air core coil assemblies. (For STS mission, the use of electromagnet assemblies was proposed.) By virtue of the wheel unloading provisions and the three-axis, active properties of the control system design, the system is essentially insensitive to the environmental disturbance profile.

4.4.3 EARTH SENSOR ASSEMBLY (ESA)

The Earth Sensor Assembly (ESA) provides spacecraft pitch and roll data to the Attitude Determination and Control subsystem. This pitch and roll data is used as an independent source of spacecraft attitude information for monitoring the Primary attitude control performance; this data is also used as the prime source of spacecraft attitude information during operation in the Backup attitude control mode.

The ESA is a static infrared horizon sensor designed to operate over an altitude range of 400 to 500 nautical miles with optimum sensitivity in the 14 to 16 micron (CO_2) band. The ESA independently views a segment of the horizon in each of 4 quadrants; 4 corresponding sets of digital horizon measurement data are obtained and provided to the on-board data processors for pitch and roll attitude determination to an accuracy of better than 0.1 degree (3 sigma). The field of view geometry of the detectors and optics was chosen to provide attitude data that is independent of levels of earth radiance.

A major design feature of the ESA is an offset radiation source (ORS) which reduces the net radiation loss to space from each of the 4 optical channels to near zero. This radiometric biasing of the sensor reduces errors arising from variations in detector responsivity and permits more accurate processing of differential radiant signals containing attitude information.

An overall simplified functional block diagram of the ESA is shown in Figure 4.4.3-1. A commutator is used to sample each of the 12 detector assembly outputs to produce a 25 Hz pulse train. A low-noise preamplifier provides the proper drive levels for the analog to digital converter. The digitized radiance signals are then formatted and provided on request to the Controls Interface Unit. The digitized radiance signals from the 4 space-viewing detectors are also fed from the A/D converter to the ORS controller, where they are used to regulate the amount of heat (radiance) generated by the ORS. Analog signal switching, A/D conversion, and formatting of the output data is under the control of the programmer.

4.4.3.1 Design Description

The ESA has 4 objective lenses which focus segments of the horizon on 4 detector assemblies as shown in Figure 4.4.3-2. Each detector assembly is equivalent to a 4 channel dc radiometer sharing a common objective lens. An individual channel consists of a 6-element thermopile detector placed behind a tetrahedron shaped optical cavity.

Exhibit 1. Description of ESA (Sheet 1 of 14)

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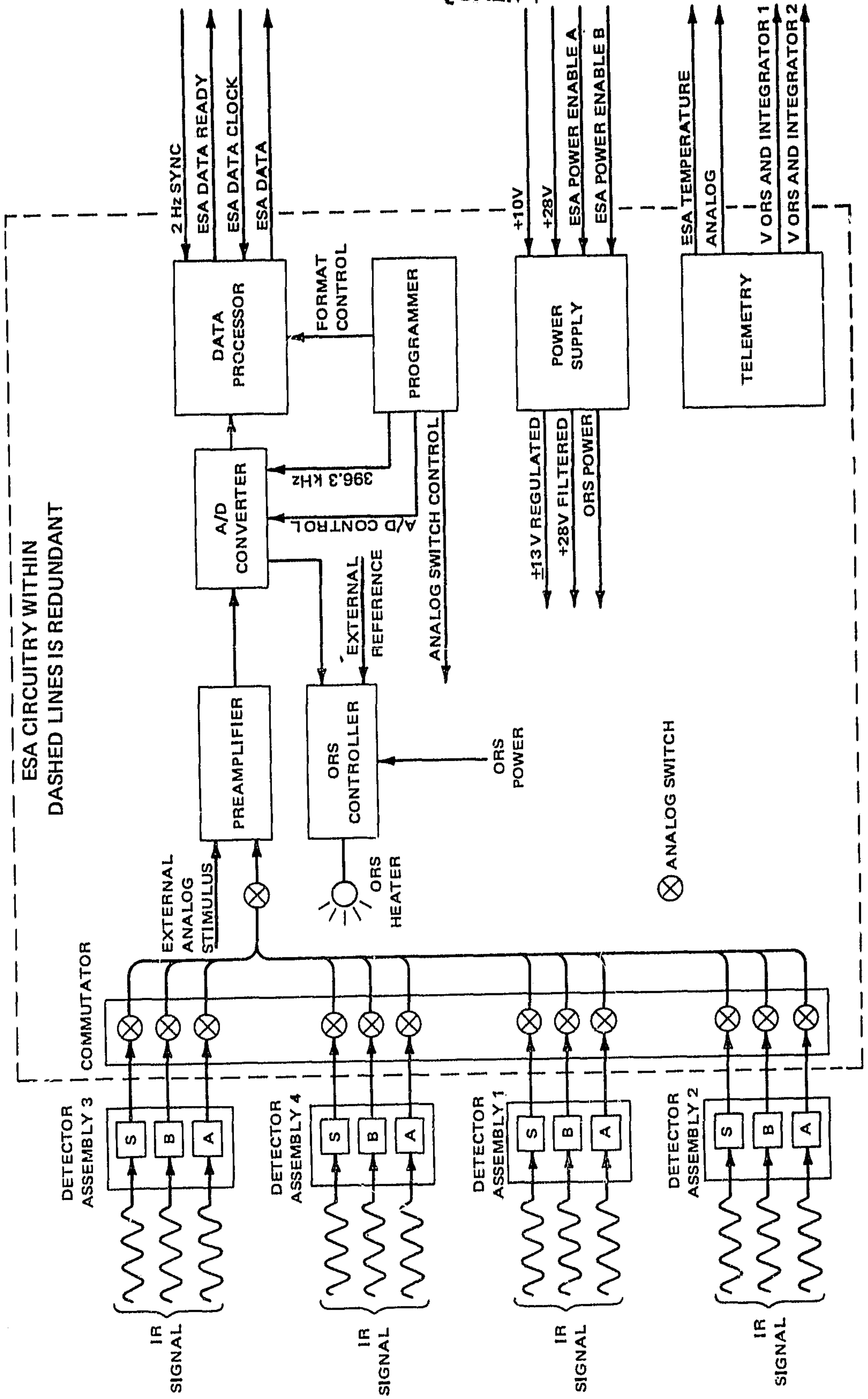

Figure 4.4.3-1. Earth Sensor Assembly Functional Block Diagram

Exhibit 1. Description of ESA (Sheet 2 of 14)

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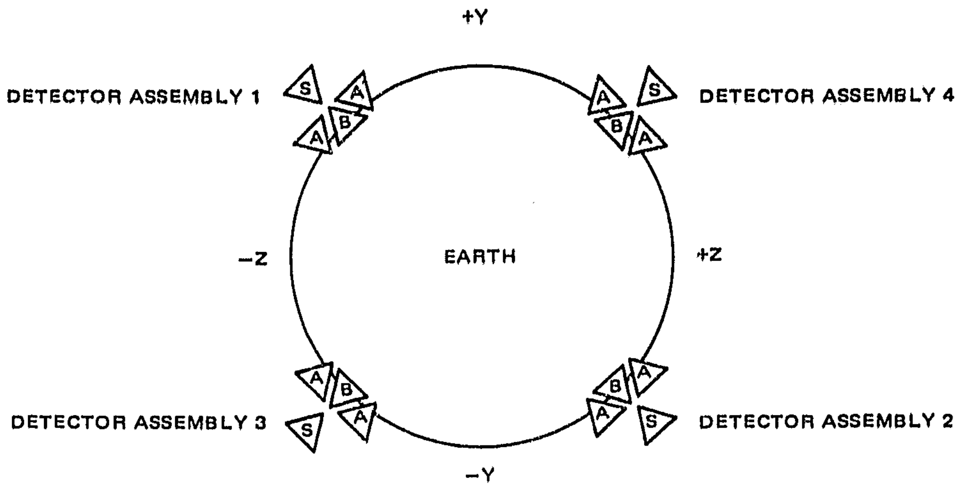

Figure 4.4.3-2. ESA Sensor Fields of View

Triangular field lenses are located at the entrance of the 4 optical cavities in each detector assembly. These lenses restrict the view angle of the detectors to the solid angle subtended by the objective. This unique optical configuration projects the 4 triangular fields of view, established by the field lenses, toward the earth's horizon. Band-pass filters are deposited on the field lenses to limit the sensitivity of each optical channel to energy in the 14 to 16 micron spectral band.

To compensate for the radiation lost to space from the detectors, an offset radiation heat source (ORS) is incorporated. The ORS heater is controlled by the most negative output from the 4 space-viewing detectors, with the heater power varied until the most negative signal is nulled. The ORS thereby provides a radiation bias to the detectors, offsetting the radiation lost to space.

Exhibit 1. Description of ESA (Sheet 3 of 14)

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TABLE 4.4.3-1. EARTH SENSOR ASSEMBLY CHARACTERISTICS

Characteristic	Value
Design Concept	Static 4-quadrant IR horizon sensor
Spectral Band	14 to 16 microns (CO_2 band)
Detector Field of View	Triangular, 6° on a side
Field of View Center	62.6° from +X axis
Accuracy (3σ at 450 nmi)	
Pitch	0.070°
Roll	0.052°
Data Output	Digital
Warm-Up Time	20 minutes
Redundancy	Dual set of electronics
Size	Approximately 7.28 diameter X 13.5 long
Weight	9.0 lbs
Power	1.35 Watt at +28 Vdc

Exhibit 1. Description of ESA (Sheet 4 of 14)

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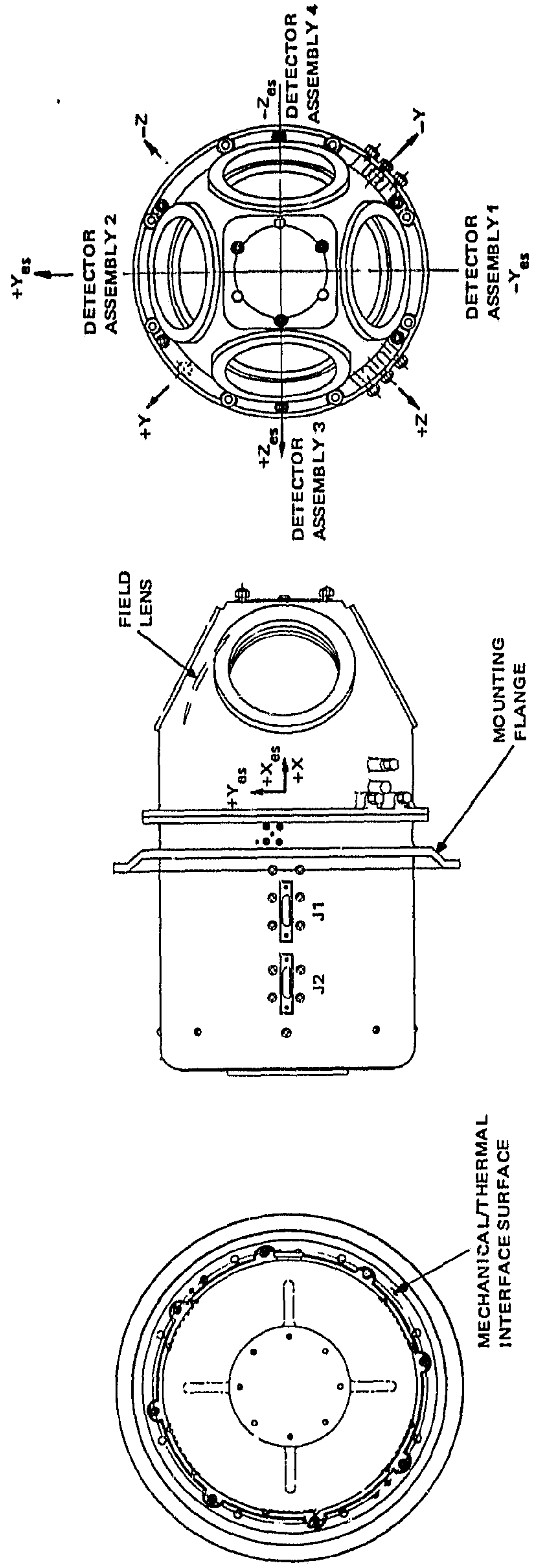

Figure 4.4.3-5. ESA External Configuration

Exhibit 1. Description of ESA (Sheet 5 of 14)

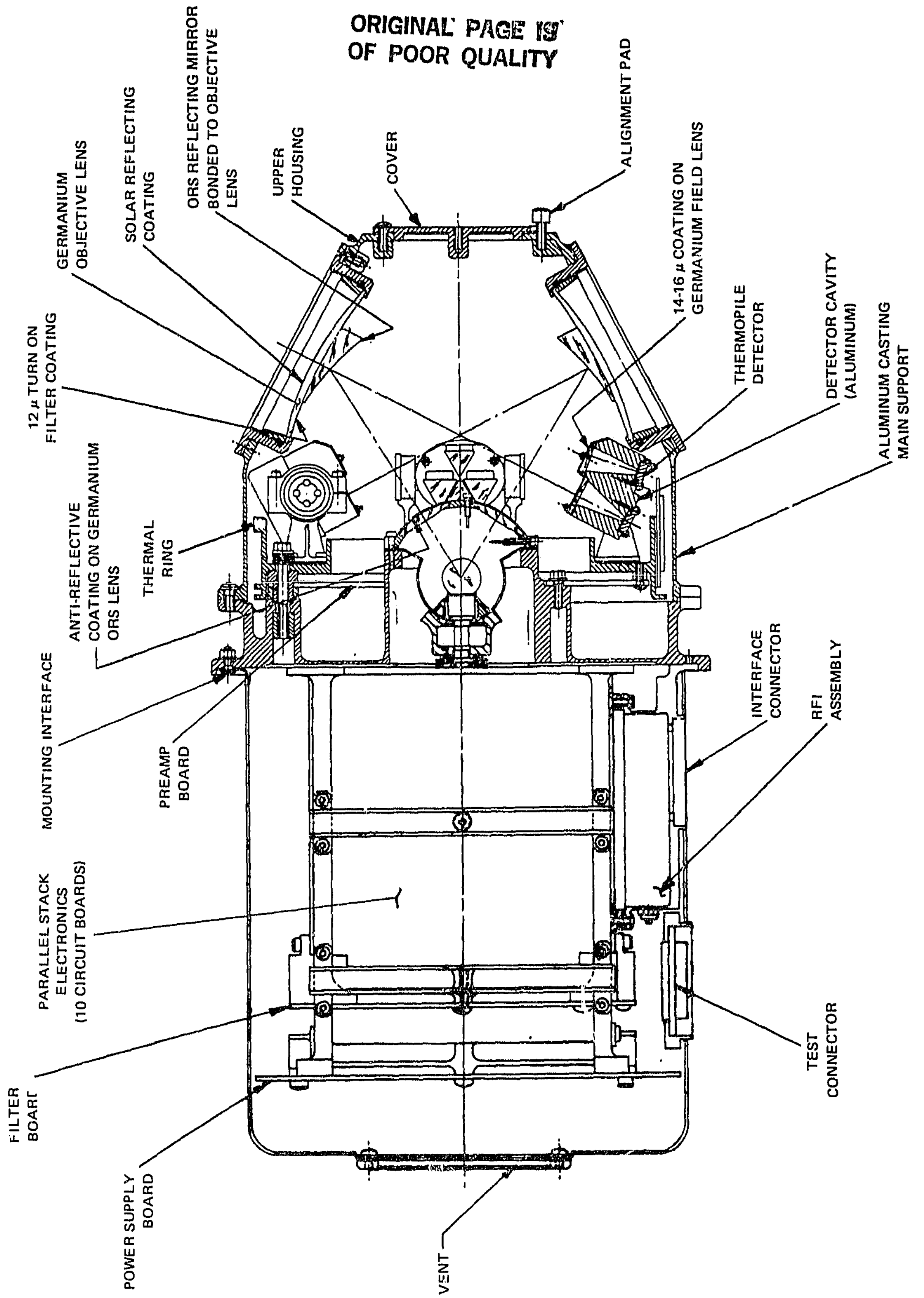

Figure 4.4.3-8. ESA Cross-Sectional Configuration

Exhibit 1. Description of ESA (Sheet 6 of 14)

C-2

TABLE 4.4.3-2. OPTICAL COMPONENT CHARACTERISTICS

Parameter	Value
Objective Lens	
Diameter	59 mm
Material	Germanium
Focal Length	f/1.7
Front Surface Coating	Reflect 80% of solar energy in 0.4 to 2.4μ spectral band
Rear Surface Coating	Reflect solar energy and block transmission in 1.8 to 12μ spectral band
Field Lens	
Size	Equilateral triangle 10.74 mm high
Material	Germanium
Clear Aperture	3.43 mm
Lens Coating	Bandpass, 14 to 16μ
Optical Cavities	
Focal Length	f/0.5
Detectors	
Type	6-Element thermopile
Active Materials	Bismuth and tellurium
Active Area	Equilateral triangle, 2.5 mm each side
Responsivity	30 volts per watt
Resistance	13,000 ohms
Time Constant	600 ms in vacuum
ORS Lens	
Diameter	20 mm
Material	Germanium

Exhibit 1. Description of ESA (Sheet 7 of 14)

TABLE 4.4.3-2. OPTICAL COMPONENT CHARACTERISTICS (Continued)

Parameter	Value
ORS Field Mirror	
Diameter	23 mm
Material	Germanium
Reflective Surface	Evaporated gold
Reflectance	>97% within 12 to 18μ band

Exhibit 1. Description of ESA (Sheet 8 of 14)

4.4.3.1.5 PERFORMANCE SUMMARY

The performance accuracy of the ESA is a function of the signal produced by an incremental attitude change, the overall ESA system noise level, and errors arising from earth radiance variations. Each of the above performance factors is discussed in the following paragraphs; Table 4.4.3-3 lists all significant sources of ESA error and summarizes its performance accuracy.

The spectral band of the ESA is determined by the filter on the lens and by the absorption of the germanium optical elements. The filter is approximately 1.7μ wide and, when combined with the rest of the optical assembly, has the spectral response shown in Figure 4.4.3-13.

The apparent temperature of the earth in the 15μ CO_2 band will be within the range of 200°K to 250°K. This spectral response has an integrated radiance of 1.026×10^{-4} W/cm^2 steradian for a 200°K earth. The clear aperture for the earth signal is 14.11 cm^2 and the signal factor is $0.86 \times 0.83 = 0.714$ (due to the cavity effective reflectivity and cavity obscuration at the detector end). The field of view (FOV) is an equilateral triangle 6.0° on an edge, or 15.59 deg^2 or $15.59 \times 0.01745^2 = 0.0047$ steradians. The signal for a 200°K apparent earth temperature is therefore 1.026×10^{-4} W/cm steradians x 14.11 cm^2 x 0.0047 x 0.714 = 4.9 μW. Figure 4.4.3-14 shows the signal as a function of apparent earth temperature.

The 6-element triangular thermopile array selected for this application has a responsivity of 30 V/W; for a fully illuminated field of view, this results in a minimum signal level of 147 μV.

The minimum signal change at null for an 0.1° attitude change will occur in the A detector for a 200°K earth at the highest altitude (500 nmi). The response of field of view A at its midpoint is 1.47 μV/5.2° = 28.3 μV/degree, giving a 2.8 μV signal change for 0.1° attitude change. For a 500 nmi altitude the horizon will actually only illuminate 1.2° of field of view A, at which point the response of A will be 1.2°/2.6° or 0.462 times its response at mid-point. The worst case signal change for 0.1° attitude change will therefore be 2.8 x 0.462 = 1.3 μV. Noise from the detector preamplifier combination has been measured to be 0.25 μV rms.

Exhibit 1. Description of ESA (Sheet 9 of 14)

TABLE 4.4.3-3. ERROR SUMMARY CHART

	3σ Performance Accuracy (Degrees)		
Significant Error Source	400 nmi	450 nmi	500 nmi
Radiance Variation			
Pitch	0.082	0.058	0.041
Roll	0.057	0.033	0.025
Responsivity Difference*	0.030	0.029	0.038
Detector and Preamplifier Noise (0.75 μV, 3σ)	0.012	0.023	0.062
Alignment	0.015	0.015	0.015
Stray Junctions (0.15 μV, 3σ)	--	--	--
Temperature Difference Between Cavities	--	--	--
Temperature Difference Between Cavities and Substrates	--	--	--
Temperature Gradient Across Objective Lens	--	--	--
RSS of Above Errors			
Pitch	0.089	0.070	0.085
Roll	0.067	0.052	0.078
*Responsivity correction to better than 1% will reduce these errors			

The ESA output, as a function of spacecraft attitude, has been optimized for an altitude of 450 nmi with an additional requirement for good linearity out to $\pm 1.0^{\circ}$ at 400 nmi and 500 nmi. These considerations led to the choice of a triangular field of view 6° on an edge, with an angle between the center of the field of view and the spacecraft +X axis of 62.60°. This field of view allows for the 2.8° required due to the variation in angle with altitude change, plus 1° additional coverage at each extreme altitude.

Exhibit 1. Description of ESA (Sheet 10 of 14)

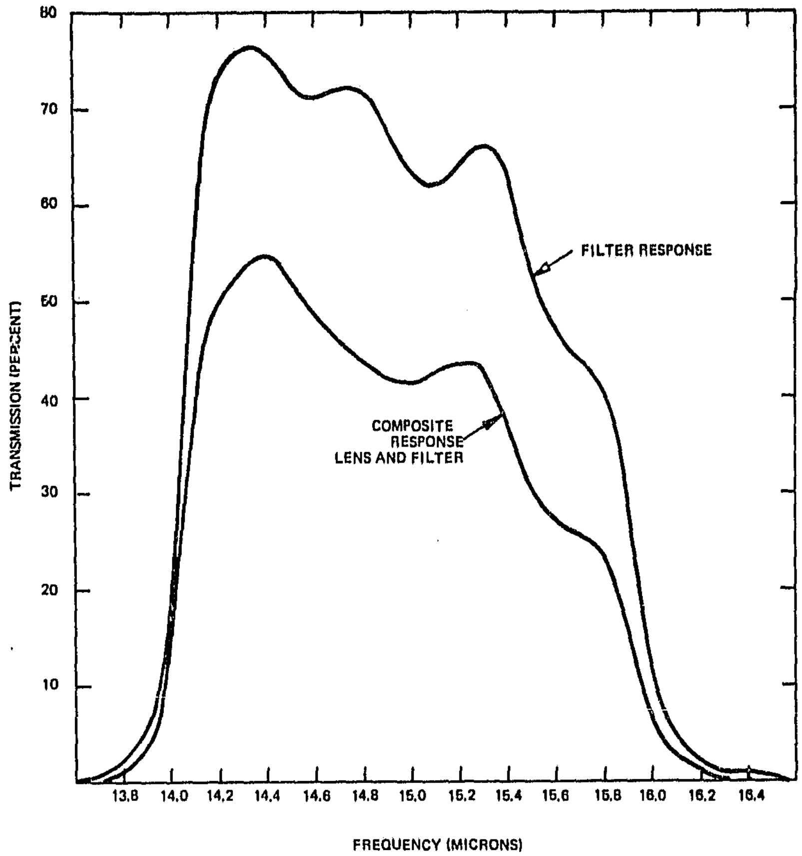

Figure 4.4.3-13. Optical Assembly Spectral Response

The system transfer functions and null errors depends upon the variation of radiance over the earth's surface; therefore, an earth model must be used to determine the anticipated performance in actual orbit. The computer program used permits the radiance at any altitude above the horizon and at all points on the earth's surface to be determined; the result is then used to compute the radiance received by fields of view A and B, from which attitude errors can also be computed.

The model used is based on horizon profiles computed from the spectral band shown in Figure 4.4.3-13 for January, April, July and October in the northern hemisphere. No reliable data exists for the southern hemisphere; it was assumed that the southern hemisphere model is the same as the northern hemisphere model of 6 months earlier. Various space measurements have indicated that radiance in the 15μ CO_2 band varies with latitude and season

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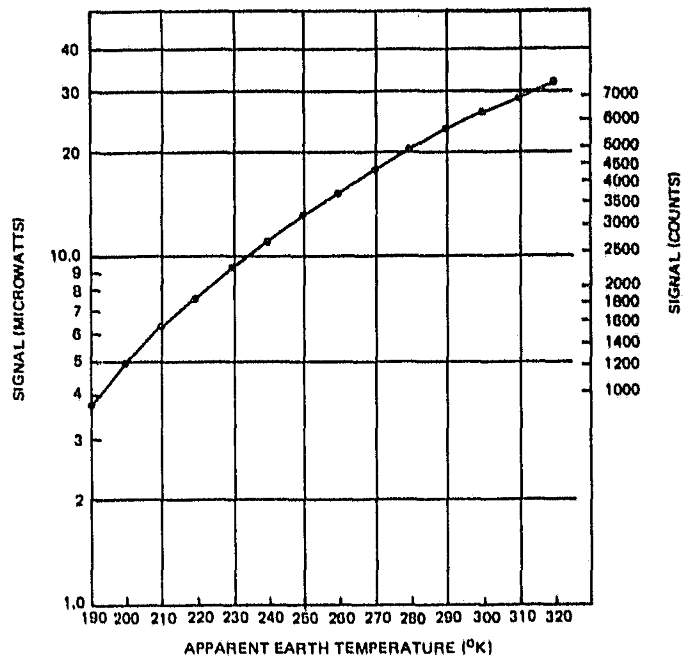

Figure 4.4.3-14. Signal vs Apparent Earth Temperature

but does not vary appreciably with longitude. The earth model therefore assumes that, since the field of view is 5.2° in the direction perpendicular to the horizon, the transfer function should theoretically be linear out to ±2.6° at 450 nmi and out to 2.6° - 1.4° or ±1.2° at the extremes in altitude, 400 nmi and 500 nmi. Actually, the thickness of the horizon itself as seen from space degrades this characteristic somewhat.

For the January orbit, the southern hemisphere was simulated by July radiance data from the northern hemisphere. For the April orbit, October northern hemisphere data was used to simulate the southern hemisphere condition. July and October orbits would therefore have error plots identical to those shown for January and April, but with a 180° phase shift.

The earth simulation computer program was used to compute pitch and roll errors due to radiance variations for 400, 450, and 500 nmi for an orbit inclination of 98.7° and for the fields of view rotated 45° with respect to the vehicle pitch and roll axes. These are shown, respectively, in Figures 4.4.3-15, 4.4.3-16, and 4.4.3-17.

Exhibit 1. Description of ESA (Sheet 12 of 14)

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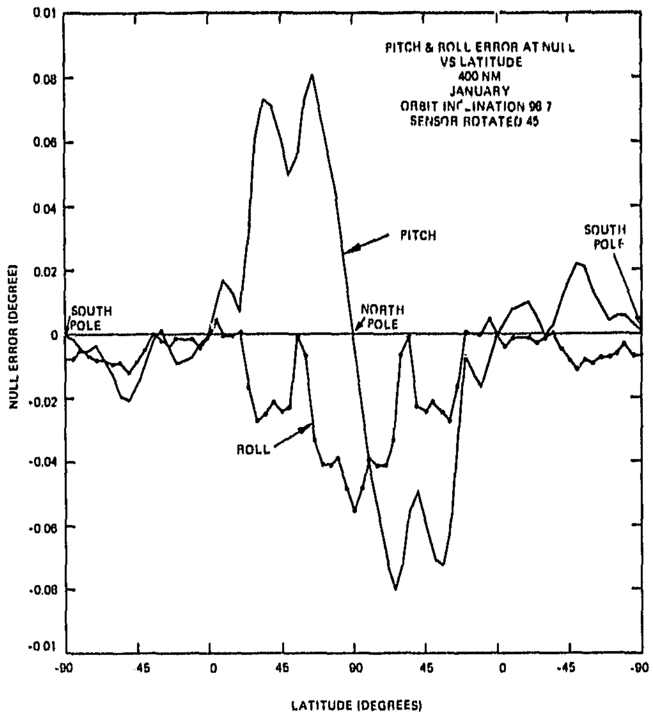

Figure 4.4.3-15. Pitch and Roll Error at Null vs Latitude (400nmi)

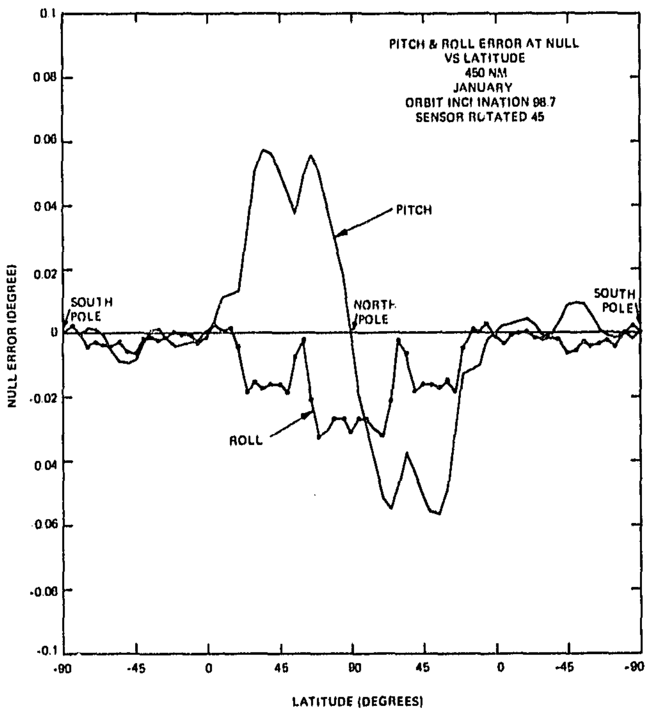

Figure 4.4.3-16. Pitch and Roll Error at Null vs Latitude (450nmi)

Exhibit 1. Description of ESA (Sheet 13 of 14)

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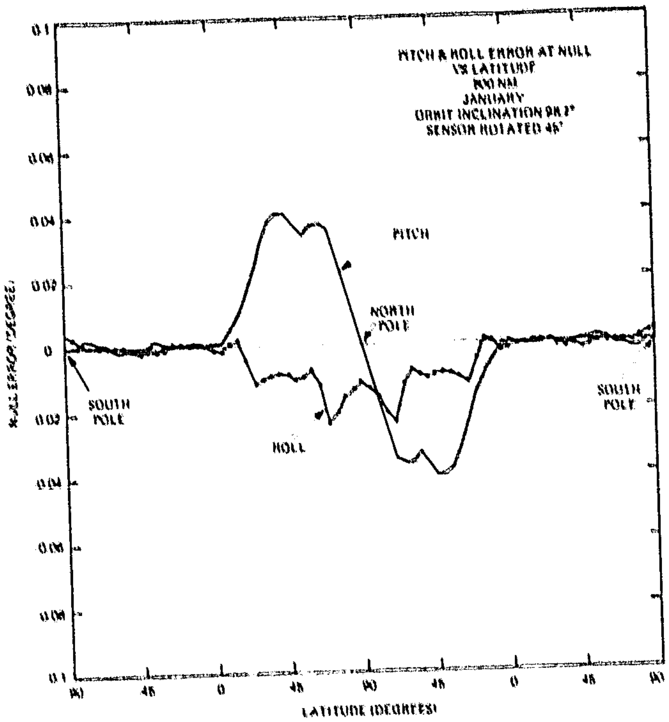

Figure 4.4.3-17. Pitch and Roll Error at Null vs Latitude (500nmi)

Exhibit 1. Description of ESA (Sheet 14 of 14)

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5.3.6.1 Conical Scanning Sensor Assembly (CSS)

Two CSS's, mounted on the Y-axis extremes of the IMP and with scan axes parallel to the Y-axis, are utilized to provide data from which the CPU can derive pitch and roll attitudes in both the nominal orbit configuration as well as the drift orbit configuration. There are two candidate suppliers of the CSS's, Barnes Engineering Company and Ithaco, Inc. To date, RCA has not completed its evaluation and made a final selection; thus, both candidate instruments are described below.

The Barnes candidate equipment to be used on each satellite consists of two sensor heads and a common electronics unit as shown in Figure 5.3-10. Sensor heads are interchangeable with electronics units. The candidate sensor is very similar to Barnes Model 13-166 horizon sensor, with customizing changes only in relatively low risk areas. Specifically, the Barnes candidate CSS

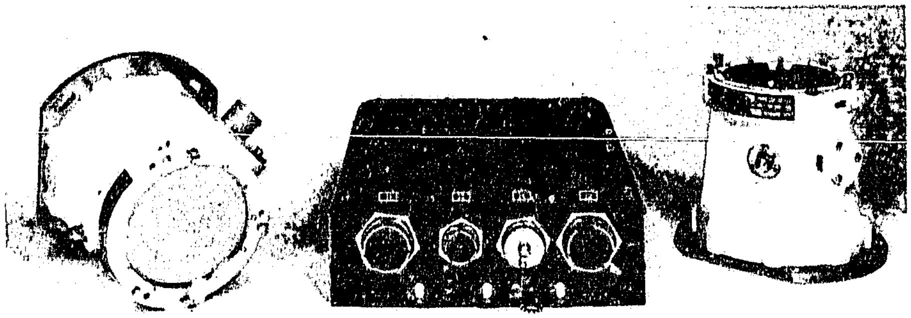

Figure 5.3-10. Barnes Model 13-166 Conical Scanning Earth Sensor

Exhibit 2. Description of CSS (Sheet 1 of 5)

shares the same hollow shaft motor/bearing and optical/detector design as the last 150 sensors of this type produced for Lockheed Missile and Space Company. Of this number, about 120 sensors have flown on the P-50 and P-95 Programs with no flight problems.

The Model 13-166 horizon sensor head has a hollow shaft motor and a modified Cassegrain optical system. The motor drives the scanning optics at a 12.25-Hz scan rate. A primary parabolic mirror, offset from the optical axis, provides a 20 degree half-cone scan. For SAATN, a prism is mounted in front of the mirror to extend the effective scan cone to a 45-degree half-cone. The secondary mirror, a hyperboloid, focuses earth radiant energy onto an immersed thermistor bolometer after being filtered by a 14-16 micrometer spectral filter. A beam splitter intercepts part of the energy before the filter and routes it to another (unimmersed) thermistor detector. This latter detector and its optics provide a 3-degree diameter field of view which is superposed over the earth sensing field of view of 2 degrees diameter. The detector is spectrally filtered to respond to thermal energy in the 2-10 micrometer region. Its function is to sense solar energy and gate out the earth channel amplifier from further signal processing when the sun is viewed. This two-color mode of operation is an effective proven method of facilitating operation without error for most sun encounters and precludes the loss of an entire scan for every sun encounter. Each head assembly is 4.6 inches deep and 4.75 inches in diameter. Mechanical modification of the 13-166 head is required to achieve the stated dimensional depth (the 13-166 depth is 5.1 inches) necessitated by the Atlas fairing envelope.

Earth signals are sensed by the immersed detector, amplified in the head, and passed on to the electronics unit for additional analog processing. Other necessary sensor processing functions are also performed digitally in a microprocessor contained within the electronics unit. Earth signals entering the 7.5 x 8.0 x 2.6 inch electronics unit are rectangular pulses which are filtered and partially differentiated. The resulting waveforms are positive-going pulses for the space-earth scan crossings and negative pulses for the earth-space crossings. In passing through an adaptive 50-percent threshold at each edge, narrow pulses are generated which define earth chord extremities. Reverence pulses, produced by a magnetic pickup in the head, provide a means of phase detecting the earth chord waveforms. Suitably formatted, this information is passed to the CPU where pitch and roll are determined. Normally, data from both CSS heads are utilized by the CPU in the attitude computations; however, data from a single head is sufficient along with data from the ephemeris. A functional block diagram of the Barnes CSS electronics is shown in Figure 5.3-11. Each system, consisting of two heads and an electronics unit, weighs 15 pounds and consumes 12 watts, steady state.

Each Ithaco CSS consists of a sensor head and an electronics box, as shown in Figure 5.3-12, and is similar to the Ithaco units being supplied for the LANDSAT D, Spacelab, and P80-1 programs. Two units (i.e. two heads and two electronics units) are flown on each satellite. The sensor head assembly contains an optical barrel with a hollow shaft dc brushless motor encircling the barrel and driving the optics at 4 Hz. The dc brushless motor is a design modification from the stepper drive utilized in previous applications of the sensor. Contained within the optical barrel are the lens and an immersed bolometer (detector). A wedge-shaped prism rotates in front of the lens to deflect the image of the detector by 45°; a coated germanium window

Exhibit 2. Description of CSS (Sheet 2 of 5)

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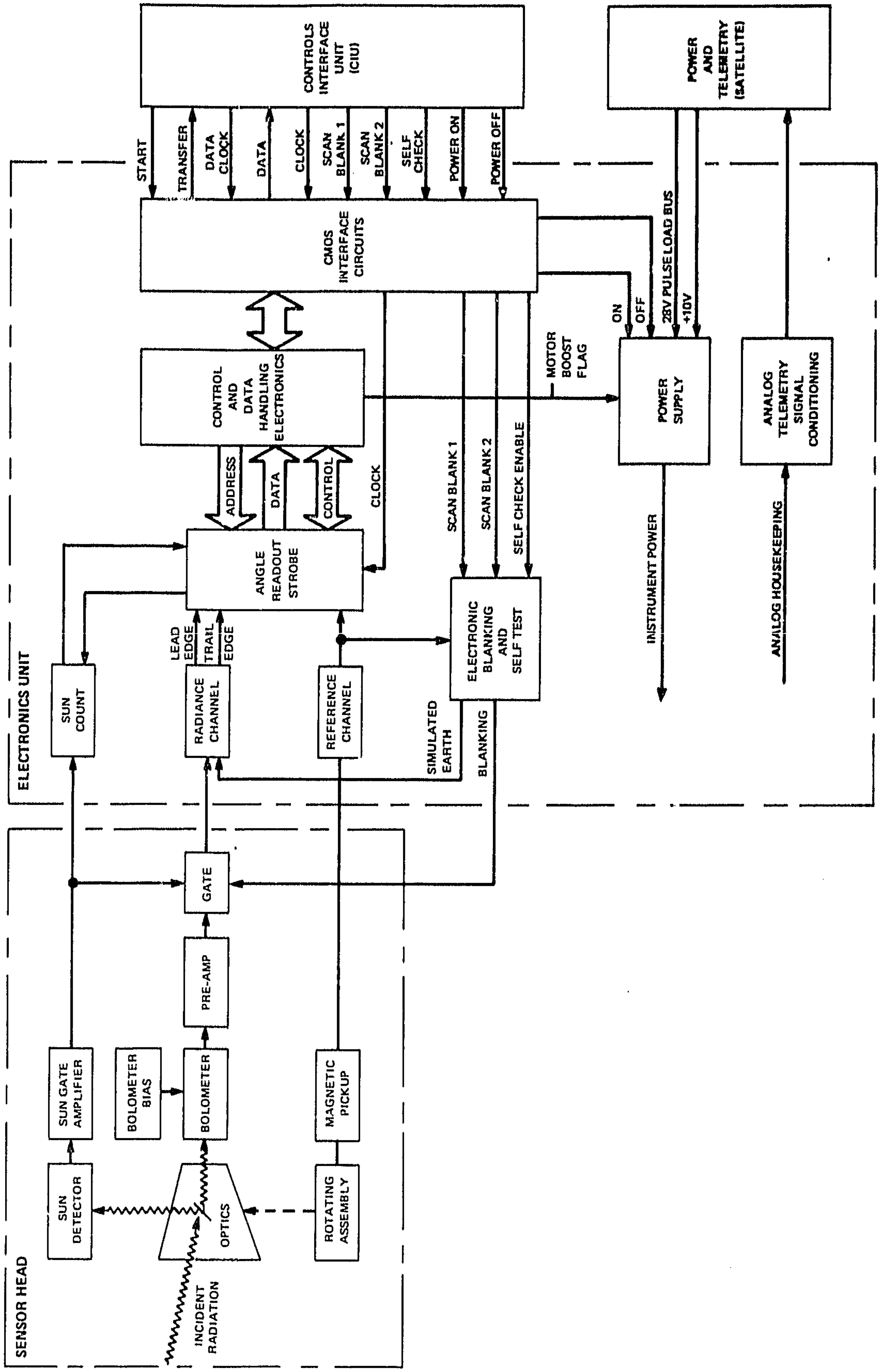

Figure 5.3-11. Barnes CSS Electronics, Block Diagram

Exhibit 2. Description of CSS (Sheet 3 of 5)

hermetically seals the unit. The optical system provides an optical bandpass between 14.0 and 16.0 microns at the half power points, with a peak response frequency at 15.0 microns and with less than 0.1 percent of the absolute energy passing below 12.5 microns and greater than 17.9 microns. As the prism rotates, the scanner field-of-view rotates in a conical motion with a 45° half-cone angle. The sensor head assembly is 4.0 inches deep and 3.0 inches in diameter.

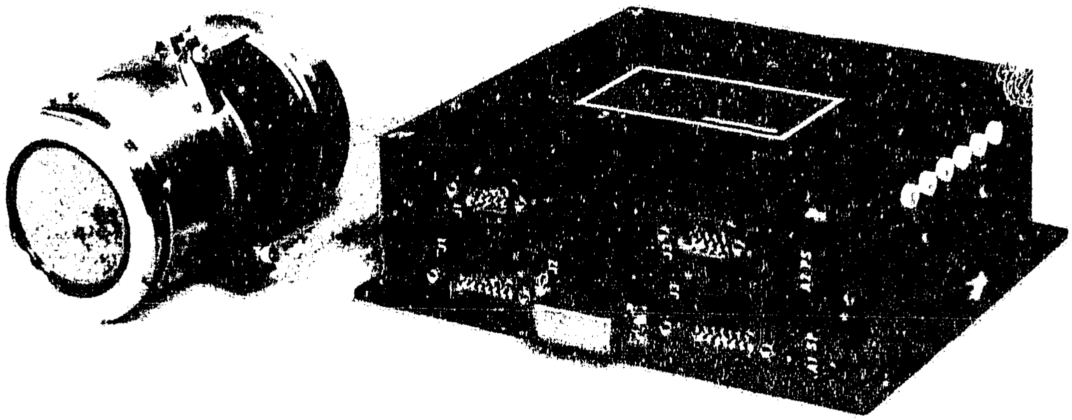

Figure 5.3-12. Ithaco Conical Scanning Earth Sensor

Once each scan, earth signals are sensed by the detector, amplified in the head and passed on to the electronics assembly for additional analog processing and digitizing of the output signals. The electronics for the sensor are contained in four modules that together make up the 6 x 8 x 4.25-inch assembly. The module functions are the power supply, motor driver, signal processor, and interface electronics. The power supply, motor driver and signal processor cards are similar to those for LANDSAT D, Spacelab, and P80-1. Changes to the instrument designs supplied on these programs are required for SAATN to provide the A/D conversion and to implement the digital interface with the Controls Interface Unit (CIU). The total weight of each CSS is less than 7.0 lb and steady-state power consumption is approximately 5.0 watts (14 lbs and 10 watts per satellite).

Earth signals, rectangular pulses which are filtered and partially differentiated, enter the electronics unit. The resulting waveforms are positive-going pulses for the space-earth scan crossings and negative pulses for the earth-space crossings. These waveforms pass through a 50-percent threshold at each edge, generating pulses which define the earth chord extremeties. Reference pulses, produced by a magnetic pickup in the head, provide the means of phase detecting the earth chord waveforms. Once each scan the sensor measures the time of the earth chord as well as the time from the sky-to-earth crossing to the phase reference pulse. This data is summed and transferred, suitably formatted, to the CIU/CPU once each half second where it is used to derive both pitch and roll. Normally, data from both CSS's are utilized by the CPU in the attitude computations; however, data from a single sensor is sufficient along with altitude data from the ephemeris.

Exhibit 2. Description of CSS (Sheet 4 of 5)

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Predicted accuracy of the pitch and roll attitude data derived from either the Barnes or the Ithaco CSS is summarized in Table 5.3-22.

TABLE 5.3-22. PREDICTED CSS ATTITUDE DETERMINATION ACCURACY

Operating Attitude (nmi)	Two-Sensor Operation Accuracy		Single-Sensor Operation Accuracy	
	Pitch (deg)	Roll (deg)	Pitch* (deg)	Roll (deg)
150	0.18	0.13	0.25	0.14
450	0.13	0.13	0.17	0.14
750	0.11	0.14	0.14	0.15
*Assumes altitude knowledge to within 1.62 nmi				

Exhibit 2. Description of CSS (Sheet 5 of 5)

QUESTION 2B: Describe the solar array pointing concept, hardware implementation, and consequences to area/mass ratio.

RESPONSE 2B: Solar array pointing is accomplished by means of a closed loop control system whose reference input is developed from an ephemeris generator module coded within the vehicles Central Processor Unit (CPU). The input is generated as an azimuth angle for the sun line projection in the plane normal to the array axis of rotation. Array rotational position developed from a shaft encoder in azimuth is fed back and compared against the reference input to develop a control error signal. The multi-mode servo controller, providing bi-directional control at one of three allowed rotational rates, processes the sensed error and develops the requisite drive input for the array stepping motor. The control is continuous, with no requirement for sun presence sensing.

QUESTION 2C: Describe the TDRS pointing concept, constraints of satellite configuration, and other pointing requirements.

RESPONSE 2C: The system under consideration uses a two-axis steerable antenna for medium and high data rate communication with TDRS. The antenna system used as a baseline is a Sperry High Gain Antenna System designed for the solar maximum mission. It uses a direct drive brushless dc gimbal control system. The input will be a set of angle commands from the CPU. The antenna will be slewed into position prior to TDRS contact using on-board ephemeris information. The antenna can be moved through a solid cone of ±110 degrees.

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QUESTION 2D: Describe the orbit correction (ΔV) maneuver and hardware implementation.

QUESTION 2D: The stationkeeping orbit correction system for the TOPEX mission is a new subsystem for the TIROS/DMSP series of satellites. The current Atlas-launched spacecraft utilize the hydrazine system not only to control attitude during solid motor(s) firing but also to make up velocity errors from the launch vehicle so that a very precise orbit is achieved. On TOPEX, the requirement for aerodynamic drag makeup in order to achieve a 10-day repeat cycle within one kilometer implies a stationkeeping function. On Page 15 of Proposal No. 103107-A, we provided a table of our estimates of ΔV requirements for each orbit altitude option. Please note that the last column was mislabled m/sec/Duty Cycle; it should have been mm/sec/Duty Cycle as shown in the next table, derived from the proposal.

Operational Alt. km (n.mi.)	Stationkeeping ΔV* m/sec/5 years (ft/sec/5 years)	Stationkeeping ΔV* mm/sec/Duty Cycle (ft/sec/Duty Cycle)
1334 (720)	Negligible	Negligible/30 days
1000 (540)	0.73 (2.37)	8.0/20 days (2.6 x 10^{-2}/20 days)
800 (430)	2.33 (7.57)	12.8/10 days (4.16 x 10^{-2}/20 days)
*Due to Air Drag Only		

Because of the location of the RCS equipment on the RSS and the location of the deployed spacecraft CG, it was necessary to add a new system. (On the Delta version, the RSS does not have any RCS equipment, and only this new system is needed.) The small total ΔV requirement and small individual ΔV corrections suggested that a simple nitrogen cold gas system would be a good candidate. Approximately 1.6 kg of nitrogen are required to meet the mission impulse requirements. In our Section 4.5 of the Final Report, we describe a blowdown system (which might be modified in future efforts to include a regulator). This system would operate from 2000 to 200 psia, with a corresponding range of thrust from 2.0 lb_f to 0.2 lb_f. The specific impulse is essentially constant at 69 seconds over the blowdown range. The shortest burn time at maximum thrust and minimum impulse bit is 5.0 seconds. The dry weight of this system is 16.8 kg, with a total nitrogen capacity of 4 kg.

With the maximum impulse bit pulse and typical misalignment of the thrust line with the CG of 0.5 inch, the attitude disturbance is less than 1.3% of the RWA capacity. Hence, the normal attitude mode can accommodate the in-plane ΔV maneuvers. As a backup on all launch vehicle options, the cold gas attitude reaction control system will automatically cut in if thresholds are exceeded.

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3. COMMAND AND DATA HANDLING

QUESTION 3A: It is unclear if the reported C&DH satisfies all of the specified TOPEX requirements (Table 2-4). For example:

a. Can a 6-kbps format be programmed for L&A and OA modes? Is a 6-kbps record mode provided?

b. Can an 8-kbps format be programmed for ALT waveform sample mode?

RESPONSE 3A: As stated in Section 4.3.2.4 of the RCA Final Report for the TOPEX Satellite Option Study, the present TIP on the TIROS/ATN program operates in either 8.32 kbps or 16.64 kbps, and changes will have to be made to accommodate TOPEX data in any case. For the study, it was assumed that the TIP data rate would be sufficient to accommodate the highest rate required (approximately 16 kbps including overhead), and the same rate would be used when lower data rates were required, with the remainder of the bits being spare or some backfill pattern provided by TIP. This should not be a problem with the link since the TOPEX system has to handle this rate at other times. Recorder capacity also should not be a problem. It is acknowledged, however, that reducing the data rate to the maximum required would reduce the TDRS SA mode access time during playback. If multiple data rates are a requirement, the modification to the TIP would include altering the format control logic to provide the proper data format for TOPEX data in more than one data rate mode. In this mode, we suggest using a common data rate for L&A, OA, and ALT waveform sample modes and a second data rate for the waveform burst mode.

QUESTION 3B: What is the RT command execution rate?

RESPONSE 3B: A command sequence consists of the command word, zero-to-three data words, and an execute word. At the CIU, each word is 25 bits long, including the active 16-bit field, spacecraft address, parity, and word type code. At 1 kbps, this results in one word per 25 milliseconds, or 40 words per second; i.e., up to 20 commands per second. Although the command verification is limited to 10 words per second, commanding can take place in bursts at a higher rate, and excess verification words are stored in a queue and output at 10 per second.

During the SAATN study, a 2-kbps STDN uplink of 32-bit words was examined. After being converted to the required 25-bit word form, the effective rate at the CIU is 62.5 words per second (as opposed to 40). The flight software is capable of handling either rate, although commanding during ascent maneuvers is not recommended.

QUESTION 3C: Does the system provide uplink parity check on commands?

RESPONSE 3C: Yes. One of the 25 bits in each uplink word is a parity check bit on that word.

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QUESTION 3D: What is the time tag accuracy?

RESPONSE 3D: There are two types of time tags: software stored command time tags and sensor data time tags supplied by TIP. The stored command time tags are compared to a software clock which is incremented by hardware interrupts traceable in accuracy to the spacecraft master clock, the RXO. Because processing of stored commands takes place at a relatively low priority level, higher level processing can make execution of stored commands differ by a few milliseconds from exact multiples of seconds.

Data time tags supplied by TIP are also derived from the RXO (master clock) and, thus, have the same stability (once initiated) as the RXO. As reported in Volume 1 of Proposal 103107-A "TOPEX Satellite Option Study," DMSP/TIROS oscillators provide stability of only 1 part in 10^8 per day. RCA's NOVA navigation satellite oscillators are specified to only 5 parts in 10^{10} per day. Individual samples of NOVA oscillators, however, have been measured at better than 5 parts in 10^{11} per day. Thus, it should be possible to test and select NOVA oscillators for TOPEX which will meet the 1 part in 10^{10} per day TOPEX requirement.

Figure 3D-1 shows how the time-code-increment clock is related to the master oscillator on the TIROS spacecraft. Thus, the stability of the time code is directly related to the stability of the master oscillator. For TOPEX, a similar scheme will be used, but with frequency division and multiplication chosen to provide the higher-resolution TOPEX time-code clock.

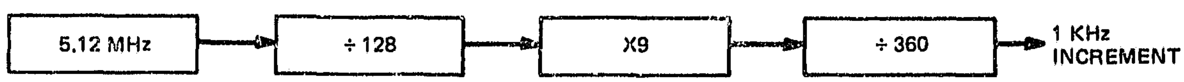

Figure 3D-1. TIROS Time Code Clock Increment

QUESTION 3E: What is the volume of C&DH?

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RESPONSE 3E:

Command

CPU	= 11.2 x 5.1 x 12.1 x 2	=	1382
CIU	= 17.6 x 7.6 x 9.3	=	1244
RXO	= 4.9 x 6.0 x 2.0	=	59
SCU	= 12.9 x 6.9 x 6.5	=	579
CPC	= Part of CIU	=	0
RAU	= Part of SCU	=	0
CRU	= Part of CIU	=	0
CDU*	= 5.6 x 7.7 x 8.8	=	379
	Subtotal	=	3643 cubic inches

Data Handling

TIP	= 14.0 x 7.6 x 8.75	=	931
XSU	= 7.56 x 6.5 x 6.5	=	319
DTR	= 11.0 x 7.0 x 9.4 x 3	=	2171
SATCU	= 7.5 x 1.9 x 0.8 (on Mast)	=	12
	Subtotal	=	3433 cubic inches

Total C&DH 7076 cubic inches

*If required

QUESTION 3F: What is the volume of the 3 NASA standard tape recorders?

RESPONSE 3F: 11.0 x 7.0 x 7.75 for the box structure = 597 cubic inches, or 11.0 x 7.0 x 9.4 including a low-height harness interconnecting the two transports and one electronics of each recorder = 724 cubic inches.

QUESTION 3G: Is 32K of memory sufficient to support both C&DH and Guidance & Control computation requirements?
What margin is left?

RESPONSE 3G: Atlas-launched TIROS and ATN spacecraft use 18K and 20K memories, respectively. The SAATN mission, which is applicable to TOPEX with regard to software sizing, was estimated to require 25.5K. Thus, 6.5K is left for margin. The 25.5K includes executive, command and control, power management, thermal control, guidance and control, and ground-use-only closed loop test modules.

A breakdown of the estimated memory requirements by subsystem for the SAATN mission is shown below. The estimates are based primarily on existing TIROS or DMSP codes, modified as necessary to function during a Shuttle Launch.

SOFTWARE MEMORY SIZE REQUIREMENTS

Software Subsystems	Function Breakdown	Atlas Size (Code & Tables)	STS Size (Code & Tables)
Executive	Sequencer	0	1285
	Orbit Executive	3603	3603
	Ascent CLT	Part of AGS	537
CLT	Orbit CLT	1440	1440
Command and Control	Command and Control	4902	4902
	Telemetry	884	1050
	Power Management	644	644
	Thermal Control	220	220
Guidance and Control	Initialize Attitude	0	2928
	Mission/Drift ADACS	4989	4989
	Geometric Functions	556	556
	Ephemeris	1007	1007
	AGS	4754	-
	SAGS	-	2178
	TOTAL	22,999	25,339

CLT = Closed Loop Test
AGS = Atlas Ascent Guidance Software
SAGS = Shuttle Ascent Guidance Software

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QUESTION 3H: What is the tape recorder power requirement?

RESPONSE 3H: Power required by the tape recorders depends on operating mode and data rate. For the TOPEX mission, each recorder will use approximately 19 watts for playback, 10.5 watts for record, 2 watts for standby, and 0.1 watt for OFF.

QUESTION 3I: What are software requirements and how are they handled in C&DH?

RESPONSE 3I: The flight software is used to process all commands except a few very basic commands such as "computer ON/OFF". As the memory is primarily RAM, software also provides control over the power subsystem and thermal subsystem during anomalies. Software is used to read attitude control sensors, process the data, and output commands to control attitude-control hardware and thrusters. The software runs in different modes during the various phases of the mission. It is present and running in some form continuously starting prior to liftoff. All required functions are present in the load package. No reloading is required. Refer to Response 3G for memory allocation.

QUESTION 3J: Do orbit altitudes in the 1000- to 1334-km range present any problems from a radiation environment standpoint for the ATN electronics?

RESPONSE 3J: As mentioned in the answer to Question 1B RCA has recently contracted for an extension on the DMSP mission to 4 years. We studied radiation effects parametrically as a function of life and orbit altitude. The following is a summary of the impact of the extended TOPEX mission at 1334 km.

Figure 3J-1 shows the dose-depth curves for the 2-year ATN mission and for a 5-year TOPEX mission (approximate) for a 720-nautical-mile orbit. Note from these curves that, for the shielding thickness greater than 100 mils of aluminum, the dose for a 5-year TOPEX mission is about 20 times greater than for a 2-year ATN mission.

On the basis of the sector analysis of DMSP Block 5D-2, the ionization dose within parts in various equipment boxes for a worst case TOPEX mission has been calculated and given in Table 3J-1. This dose calculation takes into account the device package thickness of 30 mils of aluminum and circuit-board thickness of 140 mils of aluminum. Semiconductor devices which are located on inner boards will have more shielding and, hence, the dose will be much lower. The worst case dose on any device within the spacecraft is about 1.2×10^5 rads. This implies a requirement for certain devices to be shielded or radiation hardened. We estimate the cost impact to be minor.

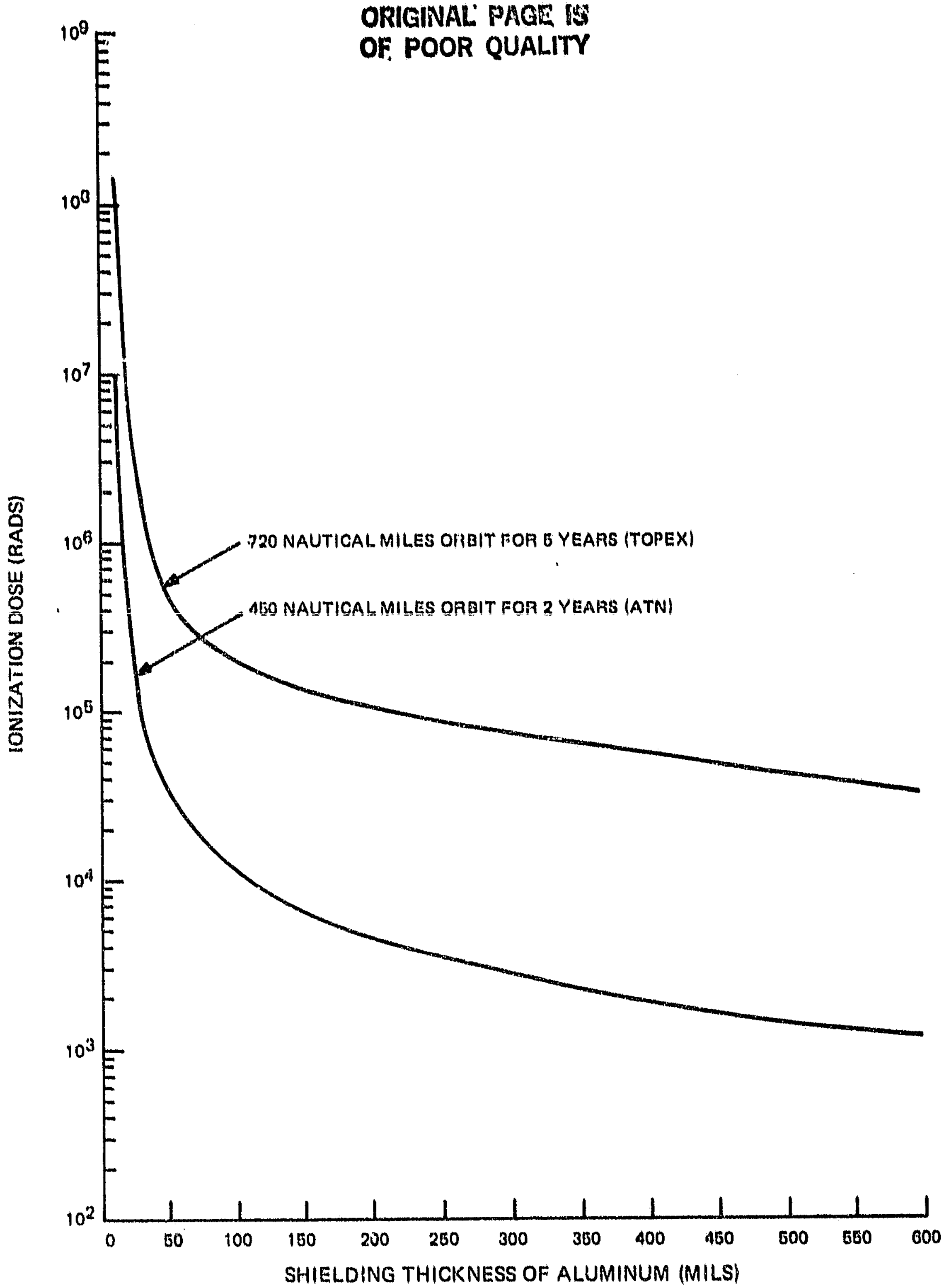

Figure 3J-1. Dose-Depth Curves for TOPEX (5-Year Mission, 720 nmi) and ATN (2-Year Mission, 450 nmi)

TABLE 3J-1. WORST CASE DOSE ON PART FOR FIVE YEARS MISSION OF TOPEX (720-NMI ORBIT)

Location of Dose Point	Ionization Dose (Rads)	Equivalent Shielding of Aluminum (Mils)
Inside ESA	1.2×10^5	200
Center of IMU	1×10^5	210
Center of CIU	9×10^4	240
Center of Top Void in RRW	6.5×10^4	350
Center of RXO	9.2×10^4	235
Center of SCU	1×10^5	210
Center of Top Void in PRW	1.2×10^5	200
Center of PC	6.5×10^4	350
Center of CPU-1	9×10^4	240
Center of CPU-2	9×10^4	240
Center of SSE	6.5×10^4	350
Center of PSE	1×10^5	210
Center of Top Void in YRW	9×10^4	240

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4. TELECOMMUNICATIONS

QUESTION 4A: What problems are anticipated interfacing the TDRS compatible telecom subsystems with the other subsystems?
Will there be a space problem?

RESPONSE 4A: No particular problems are expected interfacing with the TDRS compatible Telecommunications Subsystem. Interfaces are required with power, thermal and command, and data handling. Refer to the response to Question 4B (can the TIP drive the transponder?).

QUESTION 4B: Will the TIP drive the XPON directly or is a premodulator processor unit required? What redundancy is there?

RESPONSE 4B: It is expected that one of the TIP outputs (bi-phase or NRZ) and XSU outputs will directly drive the transmit portion of the transponder. The TIP is redundant and one side or the other is always on. The selected side drives redundant outputs to redundant units (i.e., TIP to XSU or TIP to two transponders).

QUESTION 4C: What sort of fault tolerance is there? Will one receiver be on at all times?

RESPONSE 4C: Both receivers will be on at all times, since with only one on, a failure of that receiver is a single-point failure despite the fact that a second receiver exists (it being off). As the transponders are not redundant within themselves (two are needed for redundancy), cross strapping of receiver outputs will take place in the command reformatting unit (CRU). Cross strapping in a TIROS receiver/demodulator unit (which is internally redundant) is shown in Figure 4C-1.

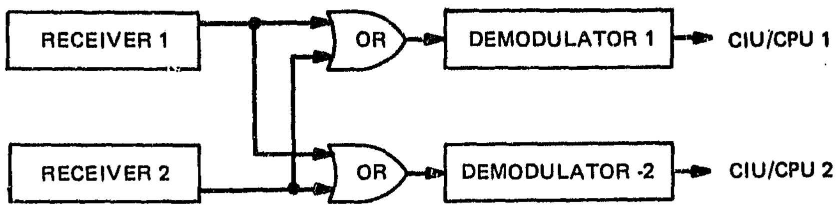

Figure 4C-1. Cross Strapping of TIROS Receiver/Demodulator Unit

The simplest method of using two nonredundant TDRS transponders is shown in Figure 4C-2. Electrical Airborne Support Equipment (EASE) is used for command when in the Shuttle. If the receiver in transponder fails, either active or inactive, that CPU is lost. Since the CPU's are redundant to each other, a loss of one CPU (although unfortunate) is not mission critical.

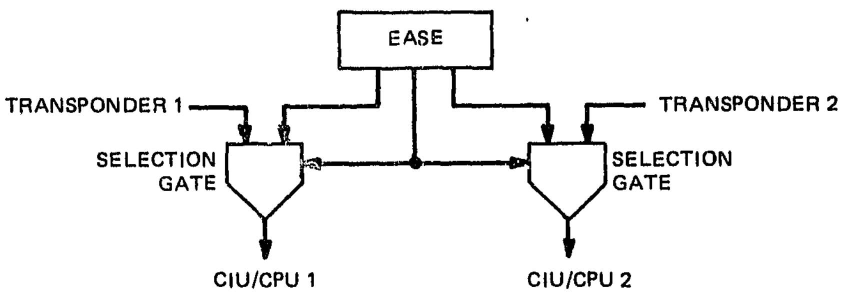

Figure 4C-2. Cross Strapping of Two Nonredundant Transponders

The approach proposed for TOPEX used a modification of the CRU cross strapping designed for SAATN. The propcsed TOPEX cross strapping is shown in Figure 4C-3.

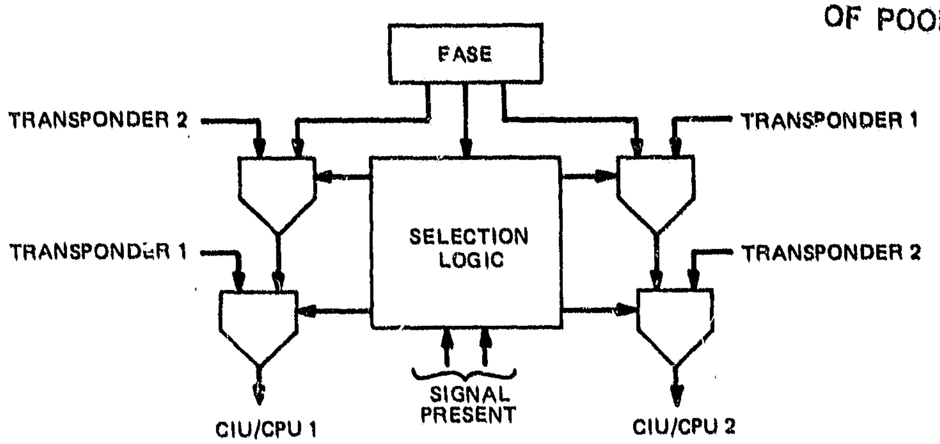

Figure 4C-3. Cross Strapping Proposed for TOPEX

The selection logic works as follows: use the EASE if the EASE is connected, otherwise select transponder 1 or 2 according to the following tabulation:

Transponder 1 Active	Yes	No	No	Yes
Transponder 2 Active	No	Yes	No	Yes
CPU 1 uses Transponder No.	1	2	2	1
CPU 2 uses Transponder No.	1	2	1	2

If a transponder somehow fails active, one CPU is lost, just as in the case of Figure 4C-2. However, if a transponder fails inactive, the remaining transponder can be used with either CPU or both.

QUESTION 4D: Please clarify which antenna serves which purpose, the gains consistent with the link tables. Figure 5.1.4-1 should reflect the clarification.

REPONSE 4D: Table 4-4 (Draft Report Table 5.1.4-3, Required Antenna Gains) is intended to show the minimum allowable gain to yield 3 dB margins for each link. Unfortunately, the tables in the draft report are not ruled. In the current Final Report, where there are options shown in a table (i.e., SA vs MA, 5-watt vs 10-watt), the selected option is boxed.

Table 4-7 (Draft Report Table 5.1.4-6) is intended to show the EIRP which would result with the minimum required antenna gain, and also the actual EIRP resulting from using one antenna for multiple links. Unfortunately some confusion arises since Table 4-6 (Draft Report Table 5.1.4-5, Typical STDN Links) was worked up for the 26-meter STDN ground antenna, whereas Table 4-7 (Draft Report Table 5.1.4-6) assumes the 9-meter STDN antenna (although this is not indicated in the table). The reason for assuming the 9-meter antenna is to allow more STDN stations to have TOPEX capability.

"TOPEX Satellite Option Study, Exhibit 1, Paragraph IV D" stipulates that an "independent S-Band link" is to be used. For this reason, antennas for the ground-direct STDN contacts are different than the TDRS antennas.

One antenna is used for beacon data at both STS and mission orbits. This is shown as "SOMA" in Figure 4-10 (Draft Report Figure 5.1.4-1) and as "SBOMA" in Table 1-2. A second antenna is used for playback of beacon data at both orbit altitudes to STDN stations. This is shown as "SHGA" in Figure 4-10 (Draft Report Figure 5.1.4-1) and "SBHGA" in Table 1-2.

For TDRS, one antenna ("TOMA" in Figure 4-10, "TDRSS OMA" in Table 1-2) is used for beacon data to TDRS. The steerable antenna is used for all other TDRSS downlinks. It is called "THGA" in Figure 4-10 and "TDRSS H-GA" in Table 1-2.

For uplink, no STDN uplink capability is shown (refer to response to Question 4H regarding STDN uplink capability). For TDRS uplink, a separate omni command antenna (-3 dB gain) is shown. The -0.3 dB gain beacon antenna (TOMA) could be used with a diplexer to feed the uplink signal to the transponder and the downlink signal to the antenna.

QUESTION 4E: How is a single-point failure avoided if the RF switches fail?

RESPONSE 4E: It is our opinion that the switches will not fail in a "no connection" mode. The switches used in the TIROS spacecraft are of a "fail-safe" design, meaning power is only required to hold the switch in one of the two positions; when this power is removed, the switch is pulled back to the other position. For the TOPEX spacecraft, we proposed that the TDRS antenna switch move to the high gain antenna position to allow data playback should a failure occur. Antenna pointing is via open loop control from the spacecraft computer based on ephemeris data. We propose that, should a failure occur, the STDN antenna switch move to the beacon antenna position to provide a backup return link from the spacecraft in the event the spacecraft cannot or does not orient the TDRS high gain antenna properly for TDRS contact.

QUESTION 4F: What HGA pointing errors and losses can be assured? Does the HGA gimbal mechanism have at least a 5-year life?

RESPONSE 4F: The HGA used as a baseline is the HGA designed for the Solar Maximum Mission. It has a pointing accuracy of 0.43 degree for fixed geometric errors and variable error sources combined. Pointing repeatability is 0.28 degree, considering variable errors only. In the SMM orbit (575 km, 33-degree inclination), the life capability of the HGA is 10 years or more.

QUESTION 4G: Should the uplink frequency be 2.10640625 rather than 2.10740625 GHz?

RESPONSE 4G: The frequency stated in the draft report (2.10740625 GHz) was taken from a document entitled "TOPEX Satellite Option Study Exhibit 1, Section IV (Satellite Requirements Summary) paragraph D (Telecommunications). The value from this source appears in the RCA Draft Report in Sections 2.5 and 5.1.4. There was no intent on the part of RCA to alter the frequency from whichever is the correct value.

QUESTION 4H: How will TOPEX be commanded (1) via TDRS using multiple access, and (2) from the ground?

RESPONSE 4H: To use an omni antenna for uplink, TDRS SA service must be used. When the HGA is properly oriented, it could be used for uplink in the MA mode as well as downlink. This option was not examined during the study.

Neither Paragraph D nor Table 5 of "TOPEX Satellite Option Study, Exhibit 1" refer to a STDN uplink, and this option was not examined during the study. From STDN 101.3, using a 1-kW transmitter and the 9-meter antenna, the EIRP at the ground would be +103.2 dBm. This EIRP, combined with the reduced path loss when compared to TDRS commanding, would yield a lower gain requirement for the spacecraft receive antenna than is necessary for receiving commands from TDRS, even in the SA mode.

QUESTION 4I: Is there a hardline out to the Telecommunications Subsystem?

RESPONSE 4I: The TIP provides hardline test outputs in both bi-phase and NRZ. These test points are associated with the data which goes to the cross-strap unit and from there to the tape recorders or transponder.

QUESTION 4J: Why does Table 5.1.4-6 list 500 kbps on the downlink in SSV orbit?

RESPONSE 4J: This mode could be used for playback of recorded data directly to a STDN station. Although there would be no instrument data in this orbit, it might be desirable to see beacon data which was generated at some point in the orbit other than over the STDN station.

QUESTION 4K: Please comment on ascent communications capability after release given SSV launch.

RESPONSE 4K: It is our intention that the command uplink and beacon downlink be usable during all phases of the mission following release from Shuttle. If the spacecraft is in the STS orbit for an appreciable amount of time (in a stabilized orientation), the playback of beacon data capability might be used (refer to the response Question 4J on why a 500 kbps mode exists in the SSV orbit).

QUESTION 4L: Does Table 5.1.4-1 account for receiver losses?

RESPONSE 4L: This question is not applicable to Table 4-2 (Draft Report Table 5.1.4-1) and probably refers to Table 4-3 (Draft Report Table 5.1.4-2), where the answer is "yes" as indicated by the 2.4-dB transponder loss on uplink signals.

5. POWER

QUESTION 5A: What are the predicted solar array degradation and its contributing elements for the basic 3-year mission and the 5-year extended mission?

RESPONSE 5A: The factors that contribute to array degradation after launch are:

a. Solar cell charged-particle loss.
b. Coverglass charged-particle loss.
c. Ultraviolet adhesive browning.

The coverglass charged-particle loss is estimated to degrade the solar cell short-circuit current by a maximum of 0.5 percent (fused silica used in place of microsheet on ATN has greater resistance to radiation damage). The coverglass adhesive browning loss from ultraviolet is estimated at 3 percent maximum and occurs entirely within the first few days after launch. Radiation damage to the solar cell is produced by the combination of electrons, protons, and alpha particles in the space environment and represents the most significant degradation of the three factors. Figure 5A-1 shows this effect in terms of the relative reduction in maximum power from the array due to solar cell charged particle loss plotted as a function of years in orbit. Two curves are drawn, one for the normal ATN orbit attitude and the second corresponding to the higher 1334-km TOPEX orbit. In assessing the effect on load power capability of the ATN power system, these curves only serve as an approximate indication of relative change, since the array seldom operates at the maximum power point on the solar cell I-V characteristic.

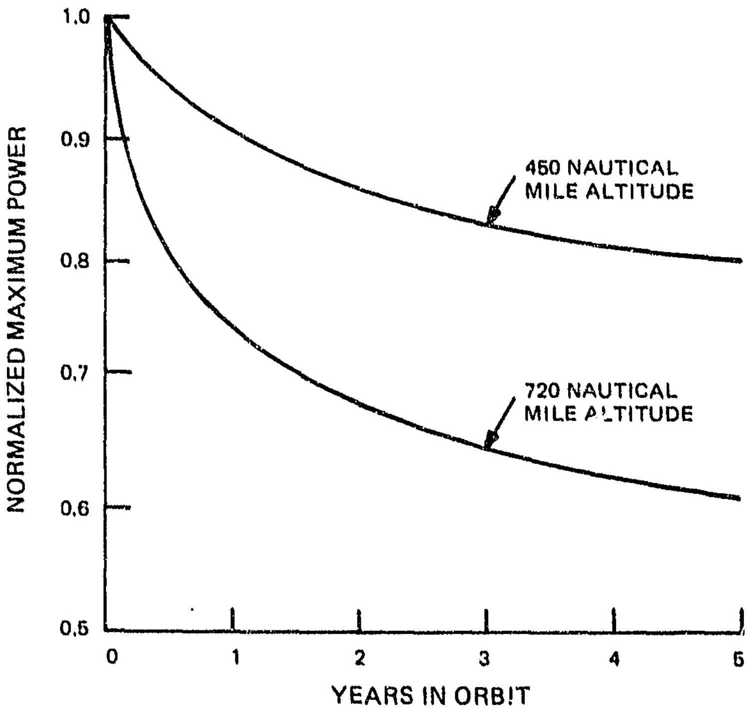

Figure 5A-1. Radiation Degradation of ATN Cell Maximum Power as a Function of Years in Orbit

QUESTION 5B: What is the predicted BOL solar array output?

RESPONSE 5B: Refer to Figure 5B-1 for a curve of maximum BOL solar array power as a function of spacecraft sun angle. This curve is derived for the ATN 450-nautical-mile sun synchronous orbit.

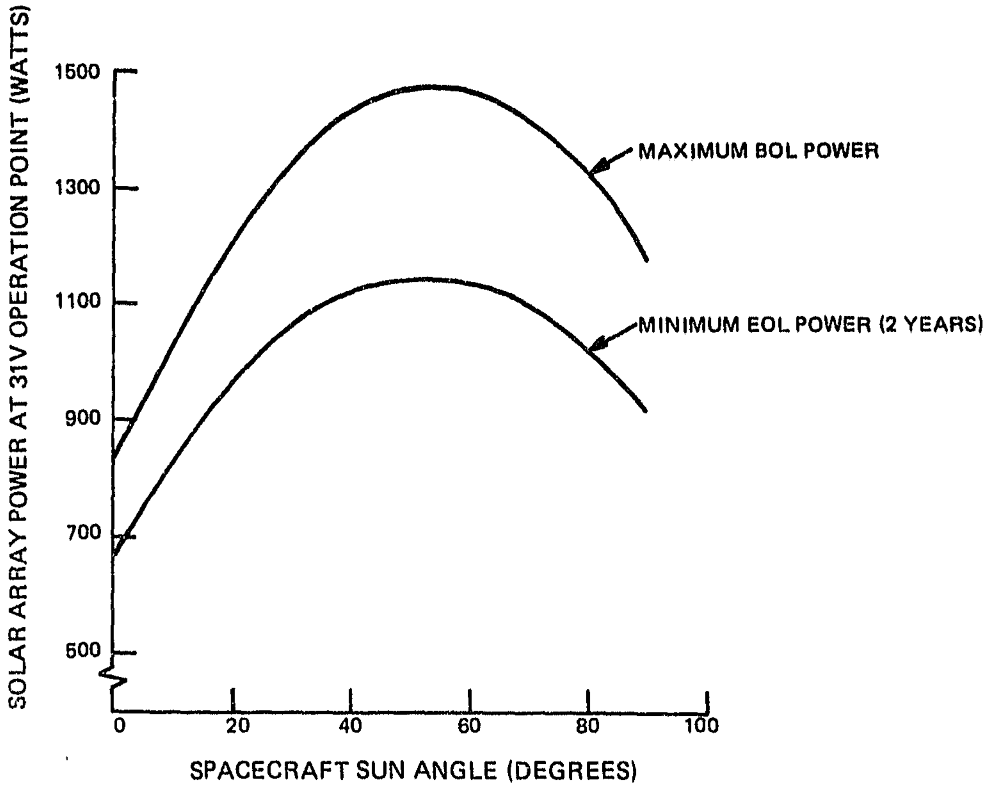

Figure 5B-1. Solar Array ATN Power Availability

QUESTION 5C: What are the tradeoff considerations for a driven cant angle system?

RESPONSE 5C: The purpose of a driven variable cant system is to maximize the array load power availability by having the capability of optimizing the array sun incidence angle for any operating orbit sun angle. However, due to sensor field-of-view considerations, as well as mechanical stability implications on the spacecraft, the allowable range of cant angles may be limited. The DMSP PMO has briefly examined this situation for their Block 5D-3 proposal and feel that they may be limited to the narrow range between 24 and 36 degrees. Because of different sensor and different jitter and pointing-accuracy requirements, the cant angle range may not be as restricted for the TOPEX series spacecraft. DMSP has proposed that a more detailed study be conducted to more accurately assess these impacts and to evaluate the tradeoff in increased load capability that can result from a variable cant array. A similar study would be required for the TOPEX mission before a recommendation can be made.

For a lower cost alternative, the use of a pre-launch presettable cant mechanism might also be considered. The preset cant angle would need to be a compromise for the range of equator crossing times possible over the life of the spacecraft, taking into account the possible launch injection errors. With the possible restrictions on the range of cant angles that may be flown coupled with the range of orbit sun angles that may have to be accommodated, the advantages of a presettable cant mechanism are not clear without additional analysis.

QUESTION 5D: What is the range of battery depth of discharge?

RESPONSE 5D: Battery depth of discharge depends on spacecraft load power and duration of night. Maximum depth of discharge would occur for the TOPEX Option 1 mission (453.8 W load) at the higher 1334 km altitude and a 90-degree sun-angle orbit (36.4-minute eclipse). For these conditions, maximum depth of discharge for a three-battery ATN power system is 19.6%. Depth of discharge becomes zero for 100% sun orbits, during which time the batteries are charged around the orbit at the trickle-charge rate.

QUESTION 5E: What is the power profile basis for "Average Load Requirements" of Table 5.1.6-1?

RESPONSE 5E: In generating the average load requirements in the referenced table, it was assumed that all of the bus components have a 100-percent duty cycle, except for the following:

Subsystem	Box	Duty Cycle	Orbit-Average Power (W)
ADACS	Roll/Yaw Torquing Coil	12%	1.7
ADACS	Torquing Coil	12%	0.7
C&DH	Digital Tape Recorder 1	100% Record	8.7
C&DH	Digital Tape Recorder 2	3 min Playback	0.5
C&DH	Digital Tape Recorder 3	100% Standby	0.1
Telecommunications	TDRS Transponder	20%	5
Telecommunications	S-Band Transmitter	20%	5

QUESTION 5F: What is the battery requirement, and how was it provided for in the solar array capability?

RESPONSE 5F: Load capability of a power system is defined as the maximum fixed load that can be continually supported by the system elements and still maintain energy balance. The batteries are said to be in energy balance if, at the end of the daylight portion of the orbit cycle, the ampere-minute charge-to-discharge rates equal or exceed the minimum requirement of 1.06 for a battery temperature of 10°C. Recharge of the batteries takes place using a modified battery voltage clamping system whereby batteries are initially recharged at a preselected constant current level until the batteries are near full state-of-charge as indicated by battery voltage. Charge current is then tapered back in order to maintain this temperature-dependent voltage as the batteries complete their recharge in the remaining daylight period. For the load capability curve of Figure 4-15 (Draft Report Figure 5.1.6-3), constant current charging is at the 7.5-ampere rate per battery, with voltage limiting at 24.99 volts (VT level 2 for a 10°C battery).

QUESTION 5G: What are the key technical problems in meeting STS Safety Requirements for power and pyro electronics?

RESPONSE 5G: The design of the basic spacecraft bus for TOPEX is based on the previously proposed SAATN spacecraft design. The STS compatible version of the SAATN design was developed to the point where a Phase 0 safety review of the spacecraft design was held with Johnson Space Center Shuttle Safety personnel. Since this review produced no safety requirement problems, the same should be true of the TOPEX design. The required number electrical lockouts for the pyrotechnics system are supplied in the spacecraft boxes which handle the firings. The remote arming board in the CIU prevents firing any pyrotechnics until the spacecraft is a safe distance from the Shuttle.

STS Safety personnel did not have any problems with the spacecraft being powered during ascent.

6. PROPULSION

QUESTION 6A: Please verify the 16.8 kg dryweight of the GN_2 thruster system and the GN_2 weight for the Delta launch option. What are the range of thrust and I_{sp} expected during blowdown operation:

RESPONSE 6A: Answered in the response to Question 2D, "Describe the orbit correction (ΔV) maneuver and hardware implementation."

QUESTION 6B: Is the isolation valve by Pyronetics listed in Table 5.1.5-2 explosively actuated?

RESPONSE 6B: Yes, this is the same valve flown on TIROS and DMSP to isolate the hydrazine tanks from the thrusters after the spacecraft transitions from orbit acquisition to orbit mode operations. (The hydrazine system is no longer required.) On TIROS/DMSP, they are normally open and fired to close; whereas, on the STS launched TOPEX, it is normally closed and fired to open.

QUESTION 6C: Should there be GN_2 isolation in the Atlas and Delta launch options?

RESPONSE 6C: On the Atlas (or Thor for DMSP 5D-1) launched TIROS/DMSP, we have not used a GN_2 isolation valve and do not plan to use one in the future. We would recommend the use of one if the mission did not require the use of the GN_2 system for an extended period of time at the start.

7. CONFIGURATION/MECHANICAL

QUESTION 7A: What is the accuracy of the 2m altimeter deployment for Delta launch?

RESPONSE 7A: The beam-pointing error on-orbit is predicted to be within 0.03 degree. This number is based on the use of deployment mechanisms and structures previously built by RCA.

QUESTION 7B: Could the ATN use the MMS cradle?

RESPONSE 7B: The use of the MMS cradle for ATN was not investigated. This would require further study by RCA.

QUESTION 7C: Are there any other designed cradles which might be considered?

RESPONSE 7C: None are known to RCA at the present time.

QUESTION 7D: Are the altimeter and radiometer pointing requirements satisfied?

RESPONSE 7D: The pointing and knowledge requirements for the TOPEX mission are different for the various payload configurations. These requirements are:

Option	Pointing	Knowledge
1	±0.15°	±0.05°
2	±0.25°	±0.10°
3	±0.25°	±0.10°

The attitude control system for the present TIROS spacecraft system has a pointing requirement of 0.20° 3σ and a knowledge requirement of 0.10° 3σ which meet the Option 2 and Option 3 TOPEX requirements. Flight data has shown that the obtained knowledge for the past three flights has been less than 0.06°, which is very close to the Option 1 requirement. Another RCA program which is very similar to TIROS (and therefore TOPEX) has a slightly different attitude control system with the following requirements: pointing ±0.01° 3σ; and knowledge ±0.005° 3σ. Future work will be performed early in the TOPEX program to determine if portions of this latter system are necessary for TOPEX.

QUESTION 7E: What is the area-to-mass ratio as apparent along the velocity vector, the sun vector, and the nadir direction for several sun angles.

RESPONSE 7E: The area-to-mass ratios are given below for each spacecraft along each of the spacecraft principal axes. The ratios are divided into the spacecraft body and solar array. The two numbers in the solar array column represent the minimum and maximum area of the array along the principal axes. As the array rotates, the area will change as a cosine function along this axis.

Configuration	AREA TO MASS RATIOS (m^2/kg)					
	X		Y		Z	
	S/C Body	Solar Array	S/C Body	Solar Array	S/C Body	Solar Array
STS 1 m	3.07×10^{-3}	6.29×10^{-5} 7.66×10^{-3}	3.05×10^{-3}	6.29×10^{-5} 7.66×10^{-3}	9.29×10^{-4}	3.65×10^{-3}
STS 2 m	4.49×10^{-3}	5.61×10^{-5} 6.83×10^{-3}	3.67×10^{-3}	5.61×10^{-5} 6.83×10^{-3}	1.91×10^{-3}	3.65×10^{-3}
Atlas 1 m	2.59×10^{-3}	4.61×10^{-5} 5.61×10^{-3}	2.57×10^{-3}	4.61×10^{-5} 5.61×10^{-3}	9.20×10^{-4}	3.65×10^{-3}
Delta 1 m	5.37×10^{-3}	9.56×10^{-5} 1.16×10^{-2}	5.34×10^{-3}	9.56×10^{-5} 1.16×10^{-2}	9.29×10^{-4}	3.65×10^{-3}
Delta 2 m	7.95×10^{-3}	9.00×10^{-5} 1.10×10^{-2}	6.65×10^{-3}	9.0×10^{-5} 1.10×10^{-2}	3.07×10^{-3}	5.85×10^{-3}

8. Cost and Programmatic

QUESTION 8A: To what extent has payload integration been considered in the Systems Engineering, Design/Development, and I&T cost elements?

RESPONSE 8A: The cost estimate for TOPEX was based on our latest estimates for the ATN and SAATN programs. The costs include payload system engineering support from the beginning of the program, for specifications on the payload as well as spacecraft accommodation, through combined test of the spacecraft bus and payload, launch, and initial orbit operations. A dedicated payload systems engineer will be responsible for interface definition, acceptance test data review, resolution of TDR's, design reviews (payload and bus), etc.

QUESTION 8B: What level of sensor manufacturer involvement is assumed?

RESPONSE 8B: Other than that which he requires to support the design fabrication and test of his sensor, his involvement is minimal. We would expect his presence during delivery and bench checkout (any special STE, targets, or handling fixtures are assumed to be provided by him), during mechanical integration and initial electrical turn-on, during major environmental test periods, during initial orbit checkout, and on-call for consultation during other periods.

QUESTION 8C: Are there any TOPEX unique requirements which if modified would reduce cost or technical risk?

RESPONSE 8C: There are several areas which could reduce cost or technical risk but which may be unacceptable programmatically. Without comment on this last problem, we list the following areas (the order of listing has no significance):

- Launch Vehicle - launch by Atlas if possible because of existing TIROS/DMSP compatibility.

  - spacecraft costs for launch by Delta is less expensive than STS.

- Altimeter Diameter - larger diameter is less risky with STS launch because no deployment is required.

- Station Keeping - this is a new system for TIROS/DMSP. Therefore, if this requirement is removed (particularly for the higher altitude options), costs could be reduced.

- TDRS Compatibility - this is a new system for TIROS/DMSP. There may be cost effective ways of operating a "bent-pipe" automated ground station that could reduce costs.

QUESTION 8D Please comment on the feasibility of using the ATNAGE hardware and/or software in a POCC environment for mission operations.

RESPONSE 8D: Although using the ATNAGE hardware and software in the POCC for mission operations has not been investigated in detail, there does not seem to be any significant technical problem with using this approach for TOPEX. There will have to be a new receiving antenna interfaced with the ATNAGE receivers and possibly higher power transmitters procurred. Hardware changes in this category should be small compared to the total system. In the software area, since the normal ATNAGE subroutines and structure are used to completely test the spacecraft, they would be adequate to use in the POCC. This does not mean that the ATNAGE software would be the most optimum for use in a POCC, but they would suffice if the procurement of software specifically designed for a POCC environment was not possible.

QUESTION 8E: Please provide cost estimates for NOAA, ATN, and DMSP missions for comparisons with TOPEX.

RESPONSE 8E: The TIROS-N program was awarded to RCA in 1975 and encompassed the design, development, fabrication, assembly, integration, test, and launch of one protoflight (TIROS-N) spacecraft, seven flight spacecraft (NOAA A-G), and all associated ground test equipment, tools and fixtures, and software necessary to support the program. Two contracts, totaling 76 million dollars, were established to perform this effort. The current TIROS-N NOAA spacecraft series comprises a protoflight spacecraft (TIROS-N), four flight spacecraft (NOAA A-D), and three flight spacecraft of Advanced TIROS-N (ATN) design, designated NOAA-E, -F and -G. Of these, four spacecraft have been launched (TIROS-N, NOAA-A, NOAA-B, NOAA-C) and four remain for future launches.

During the performance of the contract, many modifications were implemented on the program to satisfy evolving requirements. Significant changes were:

- Redesign of the last three spacecraft, which established the ATN design configuration.

- Accommodation of additions to the instrument complement:
  - Search and Rescue
  - ERBE
  - SBUV

- Design studies leading to a dual Atlas/Shuttle compatible spacecraft design.

- Parts upgrading and replacement.

- Programmatic changes extending the program to later launch dates.

As a result of these and other changes, the current value of the contracts is 126 million dollars. The evolutionary nature of the TIROS program to its current spacecraft configuration and programmatic content makes direct comparison with TIROS historical costs at the contract level misleading. The comparison is further complicated because the economic cost factors have had wide variations during the length of the program.

The TOPEX cost estimates were derived from the Shuttle/Atlas Advanced TIROS-N (SAATN) proposal submitted to NASA/GSFC early in 1981. The SAATN estimates were based upon reproducing the TIROS baseline hardware at current labor and material prices, and included the nonrecurring design costs to redesign the flight and ground equipment for replacement of obsolete or no longer available parts and components. The SAATN estimates also included nonrecurring costs for Shuttle compatibility requirements, identified as a result of extensive funded compatibility design studies. The SAATN cost baseline is more appropriate for TOPEX than TIROS history for the reasons stated. The DMSP cost history is also not an accurate data source for similar reasons; i.e., extensive programmatic changes, long program time span, and extensive configuration changes within the block. Additionally, DMSP must satisfy unique military

requirements not appropriate to civilian applications. With this background, the following SAATN costs are given for the requested comparison:

SAATN	Costs ($K)
Nonrecurring	40232
H, I, J Recurring	78146
Total Sell	118378

The costs were estimated for three complete Shuttle/Atlas compatible spacecraft, fully integrated with GFE instruments, tested, and launched. The costs also included all special Shuttle related flight equipment (i.e., the cradle, contamination shield, and electrical support hardware) required to support the spacecraft in the Shuttle bay. TOPEX cost estimates were based upon the SAATN data base adjusted to current year dollars, and included estimates for TOPEX-unique nonrecurring design costs.

9. RELIABILITY/QUALITY

QUESTION 9A: Do you have a formal reliability program in place? To what specifications?

RESPONSE 9A: Existing formal Reliability Programs are in place at RCA Astro-Electronics which are based on applicable portions of MIL-STD-1543 (USAF), MIL-STD-891A, SAMSO Exhibit 73-2B, GSFC Specification S480-4 (ATN and Block 5-D2), and RCA Document 2280925. Other formal plans are in place to support commercial communications satellites programs.

QUESTION 9B: What was the single-point failure policy for the design?

RESPONSE 9B: Single-point failure policies are defined in the internal RCA Operating Instruction 8910 when FEMA's are required by contract. The plans for the FEMA's, are incorporated in the ɩn the Reliability Program plan specifically tailored to each program. Analysis is usually accomplished to the circuit level to identify all single-point failures for either elimination or reduction to acceptable levels by compensatory techniques when possible.

QUESTION 9C: Does the design comply with current Shuttle safety requirements (such as fracture mechanics)?

RESPONSE 9C: As mentioned earlier, the spacecraft proposed for TOPEX evolved from the SAATN program. As such, all Shuttle safety requirements are met, whether they are in the areas of propulsion and pyrotechnics (previously described) or in the areas of structural factors of safety, emergency landing loads, and pressure vessel requirements. Based on the SAATN Phase 0 Safety Review, none of the STS safety requirements will require hardware changes from the proposed TOPEX spacecraft. There will, however, be many analyses and test results which must be submitted during a Shuttle spacecraft program, and these have been included.

QUESTION 9D: Do you have an electronic parts screening program in place? To what specifications?

RESPONSE 9D: Parts screening requirements are identified in the Reliability Program Plans and, when required, in a separate Parts, Materials, and Processes Program Plan. Documents are in place which use NASA/Goddard PPL-15 screening requirements, MIL-STD-891A, and SAMSO Exhibit 73-2B as a basis.

QUESTION 9E: Is there a materials and processes control program in place? To what specifications?

RESPONSE 9E: RCA Astro-Electronics has a comprehensive Parts, Materials, and Processes (PM&P) control program applicable to space flight hardware which includes specific requirements for selection, specification, application, qualification, procurement, handling, storage, and testing of PM&P. Documents which control and enforce requirements, include RCA operating instructions (policy) and program plans which are tailored to include specific program unique details. These latter documents are generated by our Reliability and Quality Engineering Groups and are subject to formal change control.

Requirements for the PM&P program are based in large part on military and NASA documents such as NASA Publication 1014 on outgassing, MIL-Q-9858A and MIL-STD-1543 on Reliability, NASA PPL-15 and MIL-STD-1546 and -1547 on Parts; all are appropriately tailored and supplemented by RCA in-house specifications.

QUESTION 9F: Is formal documentation available to support hardware qualifications?

RESPONSE 9F: Each hardware qualification is controlled by detailed qualification testing procedures specifically for each piece of hardware, subsystem, and system. All comply with environmental requirements of a specific program.

QUESTION 9G: Do you have a formal quality assurance program in place? To what specification?

RESPONSE 9G: Quality Assurance Programs are indeed in place which have been developed to meet the requirements of NASA/Goddard documents, as well as plans, meeting requirements of the USAF MIL-Q-9858A, MIL-STD-1520, and MIL-C-45662A, MIL-I-45208A with definitions in accordance with MIL-STD-109B.